NASA TECHNICAL TRANSLA"ION NASA TT F-16542

SPACE ELECTRO-JET ENGINES

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(NASA-TT-F-16542) SPACE ELECTRO-JET ENGINES N76-13200
(Transemantics, Inc., Washington, D.C.)
88 p HC $5.00 CSCL 21C
Unclas
G3/20 05317

Translation of
"Kosmichwskiye Elektro-Realtivnyye Dvigateli,"
Moscow, "Znaniye" Press (Novoye v zhizni
nauke, tekhnike. Seriya "Kosmonav-
tika, astronomiya" No. 7), 1975,
pp. 1-64.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON, D.C. 20546 OCTOBER 1975

STANDARD TITLE PAGE

1. Report No. NASA TT F-16542

2. Government Accession No.

3. Recipient's Catalog No.

4. Title and Subtitle
SPACE ELECTRO-JET ENGINES

5. Report Date October 1975

6. Performing Organization Code

7. Author(s)
A. I. Morozov and
A. P. Shubin

8. Performing Organization Report No.

10. Work Unit No.

9. Performing Organization Name and Address
Transemantics, Inc.
1901 Pennsylvania Ave. NW, Wash., DC

11. Contract or Grant No. NASw 2792

13. Type of Report and Period Covered
Translation

12. Sponsoring Agency Name and Address
National Aeronautics and Space Administration, Washington, D. C. 20546

14. Sponsoring Agency Code

15. Supplementary Notes
Translation of "Kosmichwskiye Elektro-Realtivnyye Dvigateki," Moscow, "Znaniye" Press (Novoye v zhizni nauke, tekhnike. Seriya "Kosmonavtike, astronomiya" No. 7), 1975, pp. 1-64.

16. Abstract
The book deals with the present problem of space navigation - the development of electric propulsion engines and their application to present-day space programs. These engines will determine the future of space research, the solar system, and the long term operation of earth satellites in low orbits. The book discusses the use of an electric space shuttle in future multiple-use transportation systems. The book is intended for a wide range of readers interested in problems of space technology.

17. Key Words (Selected by Author(s))

18. Distribution Statement
Unclassified. Unlimited.

19. Security Classif. (of this report) Unclassified.

20. Security Classif. (of this page) Unclassified.

21. No. of Pages 87

22. Price

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Preface

The second half of this century has been marked by remarkable progress in the conquest of space and in the mastering of its secrets. At the present time, space is being widely used to obtain and transmit the most valuable information needed by science and industry. This includes observations of the earth's surface and atmosphere, relaying of radio signals, and the study of the sun, moon, planets and other celestial bodies. /3

In a relatively short time (on the order of 10-15 years), outer space will be converted into a giant experimental laboratory where research in astrophysics, physics of plasma and elementary particles, technology and many other areas will be conducted. This process has already begun.

At the same time, industrial complexes will be formed in the space around the earth, where qualitatively new materials and systems will be created under weightless and high-vacuum conditions and will then be brought to the earth.

Thus, along with information, the earth will receive a flow of materials that will grow continuously. Finally, problems of creation of extraterrestrial power stations capable of supplying the earth with energy are being actively discussed at the present time. It is obvious that this entire process will be intimately related to progress in space engines, the requirements for which will also increase steadily. The engines should become economical as possible and convenient to operate, and should solve the most diverse problems: from acceleration of heavy spacecraft to precise orientation of a telescope or a highly directional antenna.

Among the most promising engines of the future are the electric propulsion engines (EPE), which will be discussed in this brochure. /4

What are the uses of EPE?

Contrasts of space flight. Any space flight includes a series of "active" stages involving operation of the engines, and a series of "passive" stages during which the engines are inactive and the spacecraft (SC) moves by inertia.

The chief power-consuming stage is that of insertion of the SC lifted from the surface of the planet (for example, earth), in some basic or intermediate orbit. Special booster rockets are created for this stage. Certain parameters of well-known booster rockets are shown in Table 1.

The high thrust and enormous power developed by these systems are obvious. High thrusts are required to impart to the SC a velocity not below the first escape velocity. There is an antagonism: the power of the booster rocket opposes the steadily acting force of gravity.

TABLE 1. PARAMETERS OF THE "VOSTOK" AND "SATURN-5" BOOSTER ROCKETS. The specific impulse is a very important index of a jet engine: it is equal to the time during which a gram-mass of matter ejected from the engine nozzle produces a gram-force of reactive thrust. The longer this time, the less "fuel" must be consumed to generate the required thrust force. The indicated specific impulse values pertain to oxygen[1]-kerosene fuel, which is used in engines of the first stage. [1]Ratio of the exhaust velocity to the acceleration due to gravity (9.8 m/sec^2).

Type of booster rocket	Takeoff Mass, t	Takeoff Thrust, t	Specific impulse in vacuum, sec	Total power of engines, MW
Vostok	400	510	314	15,000
Saturn 5	2760-2900	3450	290	95,000

Now, the SC together with the last stage of the booster rocket, having developed the required velocity, enter a given orbit. At once everything goes silent. The "passive" state begins, in which the system moves by inertia under weightless conditions. If we now want to change the parameters of the orbit, we no longer need high-thrust engines. The SC never falls down! All we now have to do is to exert a force measured in grams of even fractions of a gram on a multiton SC to made it start a slow but steady required maneuver.[1] This is done by merely overcoming the resistance encountered by the SC as a result of drag during motion through the rarefied atmosphere, pressure of the solar wind, attraction of the moon, and sun, etc. For a spherical satellite about 2 m in diameter at a height of 200 km, this resistance is on the order of 5 g, and at a height of 1000 km, less than 0.1 g. The marked difference between the thrusts required before and after the orbital insertion is obvious. /5

Tsiolkovskiy's formula. It may seem at first glance that the small amount of thrust necessary in a space orbit will completely eliminate the problem of engine improvement This however is not the case. The small amount of thrust is developed under very extended operation conditions sometimes measured in years,[2] in contrast to the few minutes of operation of booster rocket engines. As a result, the total impulse developed by the propulsion unit of the SC - the product of the acting force and time of operation of the propulsion unit - and the mass of the working substance consumed by the engines may be very large. The main reason for the advent of EP was the need to obtain a high exhaust velocity of the jet from the engine. We shall explain why this was necessary.

The reactive thrust force F obtained when a mass m is ejected at velocity u is equal to mu. The acceleration a acquired by an SC of mass μ will therefore be $a = \dot{m}u/\mu$. /6
Hence, the same acceleration can be imparted at both a high flow rate $\dot{m}$ and a low exhaust velocity u, and vice versa: at low flow rate, but a high exhaust velocity. Low exhaust velocities are inconvenient, since a large reserve of working substance must be available on board, so that the proportion of

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1. It is obvious that the smaller this force, the longer the maneuver should last. Actually, the time of the maneuver will not be arbitrary. There are always some limitations on the duration of a maneuver, but this is a different topic.

2. Such extended operation is necessary, for example, to maintain the required orbital parameters of a long-lived satellite.

the payload in the total takeoff weight will be small.

These considerations can be given a more convincing form when Tsiolkovskiy's fundamental formula is used. Assuming that the exhaust velocity of the jet u is constant and that external forces are absent, K. E. Tsiolkovskiy derived the following dependence of the final rocket mass μ_f on the initial mass μ_0, exhaust velocity, and velocity v achieved by the SC:

$$\frac{\mu_f}{\mu_0} = e^{-\frac{v}{u}}$$

Tsiolkovskiy's formula, derived for idealized conditions, may be applied to any flight (for example, insertion of SC in an earth orbit or flight from Earth to Mars), if v stands for a certain conventional quantity called the characteristic velocity. Estimates of characteristic velocities for certain variants of space flights are given in Table 2.

It follows from Tsiolkovskiy's formula that a sharp decrease in the final mass in comparison with the initial mass takes place when $v > u$ (see Table 2). Hence, a major requirement for the creation of relatively light (and correspondingly inexpensive) space systems is an increase in the exhaust velocity u. The entire history of the creation of modern engines has consisted mainly of efforts to produce the maximum possible effective exhaust velocity.

Capabilities of thermochemical engines. In the initial stage of development of space technology, the only possible rocket engines were thermochemical liquid or solid fuel ones, since they were the only ones that were capable of developing the thrust required for orbital insertion of an SC for a relatively small weight. In the nozzles of these engines, the combustion products of high energy mixtures are accelerated. There is s simple formula for calculating the maximum attainable exhaust for a given case: $u_{max} = \sqrt{2\chi kT/(\chi - 1)M}$. /7

TABLE 2. TYPICAL CHARACTERISTIC VELOCITIES AND RATIOS OF THE FINAL MASS OF THE ROCKET TO ITS INITIAL MASS FOR DIFFERENT SPACE FLIGHTS.

Purpose of flight	Characteristic velocity, km/sec	$\frac{\mu_f}{\mu_0}$ (u = 3 km/sec)
Insertion of SC in a circular orbit around the earth at height h = 200 km	$\gtrsim 9.5$	$\gtrsim 0.04$
Transfer of SC from orbit with h = 200 km to geostationary orbit with h = 36,000 km	4	0.26
Flight to the moon with capture in an orbit of height h = 200 km from an earth orbit h_E = 200 km	4	0.26
Flight to Mars with capture in an orbit of height h_M = 200 km from earth orbit with h_E = 200 km	5.7	0.26
Flight from earth orbit of height h_E = 200 km to planets of the Jovian group:		
a) to Jupiter with capture in an orbit of height h_J = 14,000 km	24	$3 \cdot 10^{-4}$
b) to Pluto without capture in an orbit around it	8.5	$6 \cdot 10^{-2}$

Here T is the temperature in the combustion chamber, M is the mean mass of molecules of the reaction products, χ is the specific heat ratio,[1] and k is Boltzmann's constant (1.38×10^{-23} J/deg). The exhaust velocity calculated from this formula corresponds to the engine efficiency, equal to 100%.

It is evident that the exhaust velocity is primarily determined by the initial temperature T and mean molecular mass M. This is completely understandable physically, since, as a result of molecular collisions in the nozzle, the random (thermal) energy of the molecules is converted into kinetic energy of directional motion. For the same kinetic energy, the higher the velocity, the smaller the molecular mass. Thus, when choosing a fuel for a thermochemical engine, it is necessary on the one hand to use higher energy mixtures, and on the other hand, mixtures whose reaction products have a low molecular weight. /8

Table 1 lists some of the most efficient "oxidizer-propellant" combinations, the corresponding maximum exhaust velocities of the reaction products, and the temperatures attained.

1. For monatomic gases $\chi = 5/3$, and for diatomic ones, 7/5.

TABLE 3. MAXIMUM EXHAUST VELOCITIES AND TEMPERATURES IN THE COMBUSTION CHAMBER WHEN A TWO-COMPONENT FUEL IS USED.

Oxidizer - propellant	u_{max}, km/sec	T, °K
Oxygen - gasoline	4.5	4600
Oxygen - hydrogen	5.2	4200

None of the substances exists in the solid state at the temperatures indicated in this table (the most refractory element, tungsten, melts at 3670 °K) so that the very difficult problem of engine cooling arises. As a result, and for certain other reasons, the practically attainable velocity is much lower than its theoretical limit indicated in the table (compare the data of Tables 1 and 3). Comparing the values of the attainable exhaust velocity with the above-indicated values of the characteristic velocities, we see how small the ratios u/v generally are, and hence, how small the proportion of the payload.

Thus, the chemical energy produced by the combustion of rocket fuel is insufficient to give the combustion products an exhaust velocity commensurate with typical characteristic velocities (10-50 km/sec). For this reason, giant rockets are built in which the payload amounts to only a few percent of their takeoff mass. The above-mentioned rocket "Vostok" can lift 6 t of load (including the third state of the rocket) to a geocentric orbit, and "Saturn-5" can lift no more than 135 t. The limitations of thermochemical ones, capable of developing a high exhaust velocity, be built for carrier rockets? The creation of such engines would yield a great payload gain.

However, as soon as we give up the use of an ordinary chemical engine in a booster rocket, in which the working substance (propellant and oxidizer) is simultaneously an accelerated mass and the energy carrier, we are immediately confronted with the very complex problem of creating a new powerful energy source. Let us suppose that we want to create a carrier rocket based on novel principles. Even if we assume that the entire energy produced by the source is expended in generating thrust, we find that at an exhaust velocity of 10 km/sec, a power of 50 kW is necessary to generate 1 kg of thrust, and 250 kW is necessary at an exhaust velocity of 50 km/sec. Thus, the takeoff of an SC weighing several tons from the earth's surface will require an on-board power source of enormous power, from hundreds of thousands to millions of kilowatts, and the weight of such a source must be smaller than the takeoff thrust. The creation of such high-power sources for the same weights as those of modern carrier rockets and particularly for rockets of smaller weight will require colassal efforts and a great deal of time. /9

It is therefore rather improbable that chemical engines for carrier rockets will be abandoned before the year 2000.

The propulsion units of orbiting SC are an entirely different matter. In this case, as a rule, low-thrust engines are needed whose power is measured in a few kilowatts and even watts! However, they must produce high exhaust velocities. Perhaps the most versatile and simplest engines meeting these requirements are the EP. If, for example, a hydrogen ion is accelerated by a potential difference on only 1 V, it will acquire a velocity of 15 km/sec![1] /10

The selection of an electric power source for engines is also desireable because numerous devic s installed in spacecraft require electric power. Thus, the creation of a strong electric power supply aboard a spacecraft provided with an EPE "kills two birds with one stone" by supplying electric power to both the instruments and the propulsion unit of the SC.

Characteristics of Space Electric Propulsion Units

Such a unit contains four elements (Fig. 1): the EPE as such (1), electric power sources (2), tanks with a reserve of working medium and a system for the engines (3), and a control system (4). The difference between this scheme and thermochemical propulsors amounts to the emergence of a special electric power source.

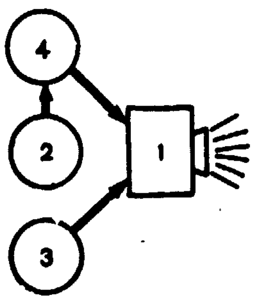

Fig. 1. Block diagram of a space electric propulsion unit: 1 - EPE; 2 - power source; 3 - working medium delivery and storage system; 4 - conversion and control system.

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1. To obtain an exhaust velocity of this magnitude thermal acceleration in a nozzle, it would be necessary to heat the hydrogen to a temperature of at least 4500°K.

This section will discuss in general terms the characteristics of only the first three units of space electric propulsion units.

Principles of EPE. Any rocket engine is based on some process of acceleration of a substance (working medium) followed by its escape. Three acceleration mechanisms can be used in EPE: the thermal, electrostatic and electromagnetic mechanisms. /11

Thermal EPE. In these engines, electrical energy is used only to heat the working medium, and its acceleration takes place in the same manner as in ordinary rocket engines, i. e., as a result of a difference in gas kinetic pressure. Two types of thermal EPE are distinguished: those heated electrically, involving indirect heating of the working medium (Fig. 2a), and electric arc engines, in which the heat source is an arc discharge fired directly in the vapor of the working medium (Fig. 2b).

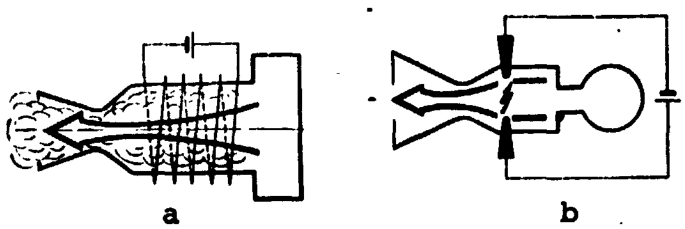

Fig. 2. Electrothermal engines: a - diagram of electrically heated propulsor; b - diagram of electric arc engine.

The capability of electric arc EPE to give an appreciable gain in exhaust velocity in comparison with ordinary thermochemical RE is fairly obvious - the arc temperature may be much higher than the temperature of the engine walls. Thus, while the maximum permissible temperature of the walls is equal to approximately 3300°K, the temperature in the arc may be several times as high. This produces an increase in exit velocity.

It may turn out that an electrically heated RE, involving indirect heating of the working medium, cannot yield a gain in exhaust velocity. Actually, this is not the case, since a number of substances (hydrogen H_2,[1] ammonia NH_3, lithium vapor and others) have a much lower molecular weight than, for example, water (H_2O) and carbon dioxide (CO_2), which are formed by combustion of propellants. It thus becomes possible /12 to create engines with a high exhaust velocity. However, this is not the only feature that designers find attractive. The exceptional simplicity of electrically heated EPE ensures a high reliability of the propulsion unit, its convenient adjustability, and the ability to withstand many activations of the engine. These are all very important advantages. Finally, in comparison with other types of EPE, electrically heated EPE require a minimum of energy consumption per unit thrust and this consumption can be reduced even further by using as a working media substances which on heating decompose with the evolution of heat (for example, hydrazine N_2H_4). As a result, electrically heated EPE, not electric-arc EPE, have now become the most important engines in the electrothermal family. An exception are low-power electrothermal pulsed engines.

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1. At a temperature of about 2500°K, the exhaust velocity of molecular hydrogen is close to 10 km/sec.

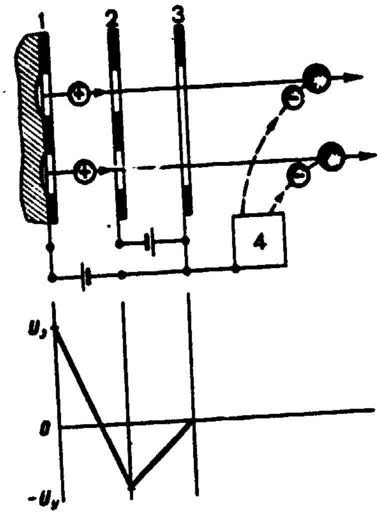

Fig. 3. Diagram of ionic engine and distribution of potential: 1 - emitter; 2 - accelerating electrode; 3 - screen; 4 - neutralizer

Electrostatic EPE. The operation of these engines involves the acceleration of charged particles of the same sign by an electric field. The most promising variants of these engines are ion engines[1], in which positively charged ions are accelerated.

The basic layout of an ion engine (Fig. 3) includ four main components: an ion source (emitter), an accelerating electrode, an outer electrode (or screen) and a compensator /13 cathode (also called neutralizer). The neutralizer and outer electrode are grounded on the frame of the SC, whose electric potential is close to the potential of space (the latter will be assumed to be zero; the emitter is at a positive po-

1. There also exist electrostatic engines in which instead of ions, fine charged droplets of liquid or dust particles (on the order of a micron or smaller) are accelerated. Such engines are called colloidal. Thus far, they are still in the preliminary developmental stage, and we shall not discuss them in detail.

tential $+U_e$, and the accelerating electrode is at a negative potential $-U_a$.

The ion engine operates as follows. The working medium in the form of a gas of vapor enters the emitter, in which atoms of the working medium ionize to form positively charged ions. The ions leave the surface of the emitter, enter the so-called acceleration gap (spacing between the emitter and the accelerating electrode), reach[2] the electric field which accelerates them, and are accelerated like a skater descending a mountain. The increase in the kinetic energy of each ion is equal to the potential difference it traverses (height of the "mountain") multiplied by the ionic charge. The accelerated ions fly across the aperture in the accelerating electrode, are decelerated to some extent (behind the accelerating electrode the ions already "rise" to a "mountain" of smaller height), move toward the screen, and after flying across its opening, are ejected outside the engine nozzle.

The electrons remaining in the emitter after ionization "go around" the electric circuit (see Fig. 3) and enter the neutralizer, which "sprays" the outgoing flux of ions with electrons. Failing this, the spacecraft would acquire an excess negative charge (the ions leave, but the electrons remain on the SC!) in the course of operation of the ion engine, and would begin to turn the ejected ions back, preventing them from leaving, from carrying their

2. Escape of the electrons is prevented by the electric field.

momentun away, and hence, from generating thrust. For this reason, it is necessary to eject the excess electrons continually. This is done by the neutralizer. Obviously, the electron current from the neutralizer should be exactly equal to the current of the ions leaving the SC.

Why then is a negative potential that does not untimately produce the acceleration of ions supplied to the accelerating electrode? It is the "mountain" obstacle to the electrons from the neutralizer (their energy is small in comparison with the energy of the ions) and does not allow them to penetrate into the accelerating gap. Otherwise, the neutralization process would be disturbed, and the electrons (whose charge is negative!) would acquire a high energy and begin to bombard the emitter. /14

In an "ideal" ion engine, all the ions leaving the emitter should fly across the opening or slit in the accelerating electrode. Hence, it is necessary not only to accelerate the ions, but also to focus the ionic flux, thus producing a suitable selected spatial distribution of the electric field in the accelerating gap.

The exhaust velocity u of ions of mass M from the ion engine is determined by the total potential difference U traversed by the ions. Obviously, $Mu^2/2 = eU$. If we are dealing with singly charged ions (the elementary charge e is equal to 1.6×10^{-19} Coul), then u (m/sec) = $10^5 \sqrt{U/50A}$, where U is the voltage in volts, and A is the atomic weight of the ion. Thus, when a 50 V potential difference is traversed, the hydrogen ions (A = 1) leave the ion engine at a velocity of 100 km/sec!

The presence in the accelerating gap of particles of the same sign leads to a peculiar self-limitation of the ion current density of the emitter by the space charge of the ions). The analogous phemomenon of limitation of the space charge of the electron current in radio tubes is well known. By virtue of this limitation, the thrust density (ratio of the thrust force to the cross sectional area of the ion beam) in an ion engine is proportional to the square of the potential difference applied.[1] In particular, in order to produce a very moderate thrust density of 0.1 g/cm^2, it is necessary to apply an electric field with an intensity of ~ 15 kV/cm. It is obvious that such a field must be created only in a narrow accelerating interval, so that the accelerating voltage is not excessively high.

<u>Electromagnetic EPE</u>. The most versatile and apparently most promising are electromagnetic rocket engines. Their operation is based on an interaction of the magnetic field with the electric current flowing across the field (Fig. 4). This interaction gives rise to an amper force. For a linear conductor with a current, this force, expressed in grams, /15

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1. More accurately, for a well-focused ion beam, $f = 4.4 \times 10^{-10} E^2$, where f is the mean thrust density (g/cm^2), and E is the mean density of the electric field (V/cm). In CGSE units, $f = E^2/8\pi$.

is:[1]

$$F = 10^{-4} IlB,$$

where I is the current in the conductor in amperes, l is its length in centimeters, and B is the magnetic field induction in gausses. We will take a current-carrying conductor 10 cm long, perpendicular to the magnetic field. If a current of 25 A flows in the conductor, and the magnetic field intensity is 200 G, the force acting on the conductor is 5 g. For a current of 5000 A and a field of 100 G, this force will be equal to 0.5 kg, and for a current of 10,000 A and a field of 1000 G, 10 kg.

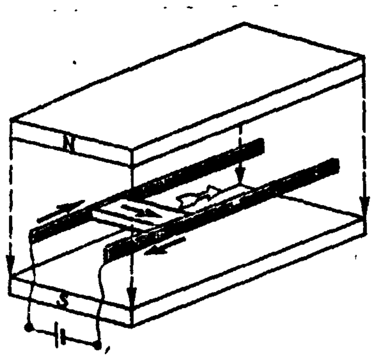

Fig. 4. Diagram of rail-type electromagnetic engine.

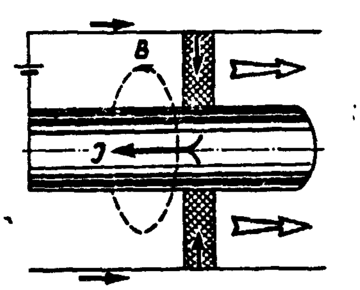

Fig. 5. Diagram of coaxial electromagnetic engine.

1. The formula is given for the most interesting case, in which the current and field are perpendicular to each other. The ampere force is directed perpendicular to the electric and magnetic field.

The exhaust velocities which the atoms (ions) of the working medium can acquire under the influence of these forces obviously depend on the mass of which these forces are acting. For the first set (I, B), the characteristic mass flow rate of the working substance $\dot{m}$ amounts to 1 mg/sec, and the exhaust velocity is around 50 km/sec. For the second set, the characteristic mass flow rate is 100 mg/sec, and the corresponding exhaust velocity is also around 50 km/sec. Similar estimates may be made for the third case. /16

The above scheme of electromagnetic acceleration is essentially a scheme of a pulsed engine with an external magnetic field, which periodically removes current-carrying connectors. Obviously, it is unrealistic to try to provide for an average flow rate on the order of 1 mg/sec at an exhaust velocity of 2-050 km/sec by means of rigid (for example, metallic) connectors. Actually, such a connector should consist of a conducting gas, i. e., a plasma containing a more or less large number of charged particles (ions and electrons). In many plasma EPE, the degree of plasma ionization is close to 100%. Thus, rigid connectors should be replaced by plasma ones. For this reason, electromagnetic EPE are always plasma ones.

Electromagnetic engines can be not only pulsed, but stationary. For this purpose, it is necessary to set up a stationary electrical supply and a continuous supply of the working medium to the interelectrode gap. The majority of electromagnetic engines are indeed of stationary type. One can also give up the external magnetic field if a large current (on the order of 10 kA or more) is passed through the plasma connector. In this case, the intrinsic magnetic field of the flowing current will be measured in hundreds and thousands of gausses and may conpletely replace the external field.

Plasma accelerators with an intrinsic magnetic field are usually given a coaxial shape (Fig. 5). In a coaxial accelerator, the current flowing through the center electrode generates an azimuthal magnetic field which, by interacting with the radial electric current in the plasma, generates an ampere force that disperses the plasma connector. This is how stationary high-current end-type and pulsed plasms engines operate.

It can be shown that in electromagnetic engines with an intrinsic magnetic field, the thrust density is proportional to the square of the magnetic field intensity:[1] $f = 4 \times 10^{-5} B^2$, so that in order to generate a thrust density of 0.1 g/cm^2, it /17 is necessary to produce a magnetic field with an intensity of only 50 G. The equivalent electric field in ion engines is equal to 15 kV/cm! It is not surprising, therefore, that the thrust density in electromagnetic engines may be much higher

1. In the CGSE system, $F = B^2/8\pi$.

than in electrostatic ones.[2]

We shall note in conclusion that in addition to the above classification of EPE according to the acceleration principle (thermal, electrostatic and electromagnetic), it is sometimes convenient to use a classification according to the state of the substance in the propulsor channel. We then have gasdynamic (electrically heated), ionic, and plasma (electric-arc and electromagnetic) propulsors.

Energy characteristics of EPE. In analyzing the operation of EPE, in addition to the magnitudes of the thrust and exhaust velocity (or specific impulse), an important role is played by the energy parameters of the EPE.

We shall consider the energy balance of EPE. The power P_F consumed in the generation of thrust can be calculated as follows. To each atom of mass M, ejected at a mean velocity u, we shall attribute a "mean" kinetic energy $Mu^2/2$. Assuming the operation of the EPE to be stationary, and multiplying this quantity by the number of atoms ejected per second, we obtain the desired relation: $P_f = \dot{m}u^2/2 = fu/2$. Separating P_F into the total electric power P supplied to the EPE, we obtain a quantity called the "thrust efficiency," equal to the ratio of the power P_j carried away by the jet to the total consumed power: $\eta_F = Fu/2p$.

Another important index is the "energy efficiency," equal to the ratio of the power P_j carried away by the jet to the total consumed power: $\eta_E = P_j/P$. It is easy to see that the thrust efficiency η_F is always smaller than atoms of the escaping jet consists of both the energy of directed motion and thermal energy. As a result, not all the particles of the working medium have a velocity that is the same in magnitude and direction at the exit.

One of the most important characteristics of EPE is the thrust rating γ, defined as the ratio of the power supplied P to the thrust force F: γ = P?F. The thrust rating shows how much power the EPE comsumes in generating a unit of thrust force. The smaller this quantity, the smaller the energy that must be "paid" for generating the thrust. The quantity γ may be written in a different form: $\gamma = U/2\eta_F$. Thus, at constant η_F, the thrust force increases with rising exhaust velocity.

In pulsed engines, the characteristic quantities used are the time-averaged mass flow rate $\bar{\dot{m}}$, thrust force $\bar{F}$ exhaust velocity $\bar{u}$, equal to the ratio $\bar{F}/\bar{\dot{m}}$, etc. The energy efficiency is defined as the ratio of the energy introduced into the jet per pulse to the total energy introduced per pulse.

<u>Supplying EPE with power and with the working medium</u>. At the present time, the chief source of electric power on board SC are solar photocells. As recently as a year or two ago, the power of SC solar batteries did not exceed 1-2kW. However, in 1973, the Skylab space laboratory, whose solar batteries produced a power level of about 15 kW, was placed in an earth orbit.

The main component of solar batteries are silicon semiconductor photocells. Individual photocells are collected into a panel. The efficiency of standard solar photopanels is around 10%. Since a solar flux with a power of 1.3 kW/m^2 strikes a panel with photocells near the earth, 100 to 150 W of electric power can be obtained from a square meter of panel. This power can indeed be obtained by accurately orienting the batteries at right angles to the direction of propagation of solar rays, and only at the outset; 6-12 months later, the "aging" of the photocells causes the efficiency to drop 5-6%, and at that point the latter may remain unchanged for several years.

Thus, the generation of 1 kW of electric power requires about 20 m^2 of panels, which weigh about 100 kg[1]. At the present time, very intensive research is being conducted to improve photocells from the standpoint of both increasing their efficiency and decreasing the weight of the panels. Already available in laboratories are samples of photocells with an efficiency of 16-18%! Success has also been achieved in the area of weight reduction by decreasing the thickness of the photocells and panels. Photopanels in the form of thin films /19

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1. Added to this weight is that of buffer storage batteries, which are required for low-flying satallites entering the earth's shadow, and also the weight of battery orientation systems. Electricity is supplied by storage batteries, which are charged by solar batteries while the satallite is illuminated by the sun and thus act as a "buffer" which smooths out the irregularity of electric power generation aboard the satellite.

less than 0.1 mm thick are being developed. The weight of such a panel, producing 1 kW of electric power, has been reduced to 15 kg and continues to decrease. It is natural to expect that in the near future, the power available to an SC will sharply increase, and a power of on-board sources amounting to several kilowatts will become the standard equipment of serially produced satellites. Although highly promising, solar power engineering is unable to solve a whole series of problems that may confront an SC with electric propulsion units. Such problems include flights on SC in low orbits, where the drag is high, future piloted interplanetary flights in ships with engines of high power measured in many thousands of kilowatts, and flights to distant planets of the Solar System (Jupiter and beyond), where the density of solar energy is very low. Therefore, in addition to photocells, considerable attention is being given to the creation of a space nuclear power plant. This constitutes a much more compact and autonomous power source for SC. Radioisotope generators have been designed and built using atomic reactors in which the heat evolved by nuclear reactions is converted by thermoelectric cells into electrical energy. However, these are comparatively low-power and heavy devices of low efficiency.

The next question arising in connection with the functioning of space electric propulsion units is that of providing EPE with a working medium. At the present time and

and in the near future, the working medium will have to be carried along from earth. Later on, as a result of the already-initiated development of space technology, the working medium will probably be obtained directly in space by reprocessing old SC, the material of asteroids, and lunar and planetary soils. The chemical nature of the working medium for EPE is not very essential: it only acts as an accelerated inert mass. In comparison with LRE, supplying the working medium to the EPE has its own specific features related to both the characteristics of the working media and a low consumption. Some of the methods currently employed will be mentioned below. /20

Modern Types of EPE

To date, a large number of different versions of EPE have been described in the literature. However, as we have noted, there are only three basic types, electrically heated, ionic and plasma EPE.

Discussed here will be some specific examples of each type of engine, but before considering them, we shall briefly review the history of EPE.

Brief History. The idea of the use of electricity for the creation of reactive thrust was first advanced by K. E. Tsiolkovskiy in his article entitled "Study of Outer Space by means of Reactive Instruments," published in 1911. Tsiolkovskiy wrote:

"Perhaps by using electricity it will be possible in time to impart a great velocity to particles ejected from a reactive instrument. We now know that the cathode rays in a Crookes tube, as well as radio beams, are associated with a flux of electrons, the mass of each of which, as we have stated, is[1] 1/4000 the mass of the helium atom, and whose velocity attains 30-100 thousand km/sec, i. e., is 6-20 thousand times the velocity of ordinary combustion products escaping from our reaction tube."

The idea of EPE was thus formulated.

Serious attention was given to electric propulsion engines by Herman Oberth, the eminent German rocket engineer. In his book, "The Road to Outer Space," published in 1929, a whole chapter was devoted to EPE. Oberth first pointed out that the thrust thus obtained will be very small, but if the engine operates for a sufficiently long period of time, the rocket can be accelerated to high velocities. /21

In 1929, the world's first construction work aimed at building EPE was started in Leningrad. Soviet scientist, V. P. Glushko, a founder of Soviet rocket engine construction, now an academician, proposed and investigated the first pulsed electrothermal engine, in which the working substance, supplied to the working chamber in portions, either as a solid or as a liquid, was heated to very high temperatures by means of an electric explosion, which was carried out by passing a powerful electrical discharge

1. Actually, approximately 1/7500.

through the working substance by means of a battery of capacitors, whereupon the working substance escaped through a nozzle (Fig. 6).

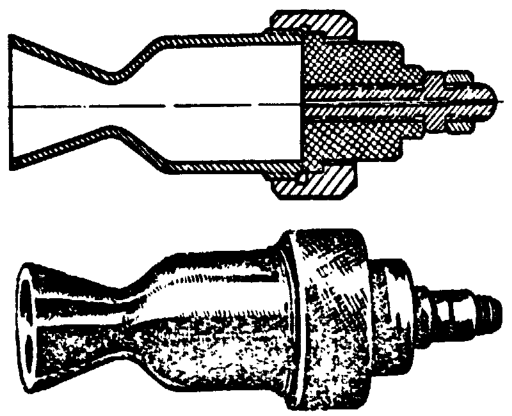

Fig. 6. Electrothermal engine of V. P. Glushko's design.

An intensive development of EPE began, however, only in the middle 1950's. A powerful impetus for this development was provided, on the one hand, by the development in connection with the problem of creation of controlled thermonuclear fusion, of a new area of physics, i. e., high-temperature plasma physics. Thus, first in the Soviet Union, and then in the USA, electromagnetic "rail-type"[1] then coaxial plasma injectors appeared (see Fig. 4), in which the plasma blobs acquired a velocity of up to 100 km/sec. /22

The eminent Soviet physicists I. V. Kurchatov and L. A. Artsimovich, who led the research in plasma physics in the USSR, have seriously turned their attention to the development of research in EPE. S. P. Korolev lent his enthusias-

1. In 1957, an article by L. A. Artsimovich, S. Yu. Luk'yanov, I. M. Podgornyy, and S. A. Chuvatin, which first described experiments on the electrodynamic acceleration of plasma blobs, was published.

tic support to this research.

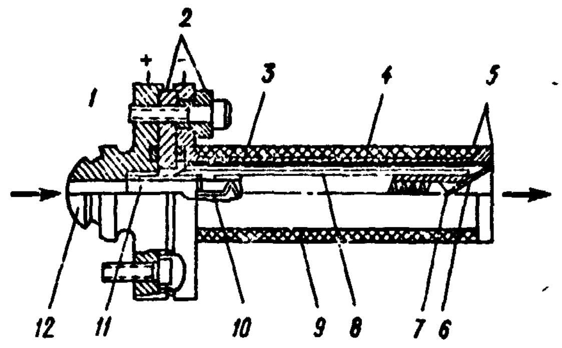

Fig. 7. Electrically heated engine with spiral heater: 1 - seals; 2 - insulators; 3 - vacuum; 4 - outer casing; 5 - electron beam welding; 6 - rhenium nozzle; 7 - ceramic support; 8 - rhenium tube; 9 - insulation; 10 - rhenium spiral of heater; 11 - electrical connection and compensation of thermal expansion; 12 - guiding flange.

The favorable atmosphere created for EPE research in the Soviet Union rapidly bore fruit. In 1964, Soviet pulsed EPE were sent out into space. In the Soviet Union, space tests of ion (1966) and stationary plasma (1972) engines were carried out. Intensive space testing of various types of EPE also began in the USA. This will be discussed in more detail in the next section.

Electrically heated engines. The basic layout of these engines was described above. At the present time, electrically heated EPE, called "resistojet" in the foreign literature, /23 are being designed for a power from a few watts to several kilowatts for purposes of spacecraft attitude control and correction.

Figure 7 shows an electrically heated engine with a spiral shenium heater, designed for a thrust of 9 g and operating on ammonia. The heater operated at a temperature of about 1000°K. In this engine, the specific impulse is 200 sec, the thrust rating is only 15.4 W/g, and the thrust efficiency reaches 52%. The diameter of the critical nozzle cross section of an engine made of rhenium is 0.76 mm.

At the present time, more powerful electrically heated EPE with a high heating temperature (about 2500°K) are being designed. A model of an engine with a high thrust of 66 g and a specific impulse of 800 sec, consuming a power of 3 kW (thrust rating, 45.5 W/g) and operating on hydrogen, has been designed and tested in the laboratory.

Although the specific impulse of an electrically heated EPE may be higher than in LRE, it is still considerably below that of other types of EPE. The exceptionally low, in comparison with other EPE, thrust rating of these engines may in many cases be of decisive importance in the selection of this type of EPE for comparatively short-lived spacecraft if the energy reserve on board the SC is very limited.

Ion engines. Ionization of substance. While the substance in the region of acceleration (nozzle) in an electrically heated EPE remains unionized, the situation is entirely different in ion engines. In this case, the acceleration zone contains only electrically charged particles, i. e., ions of the same sign (see Fig. 3). Existing ion engines differ mainly in the method of "preparation" of the ions, i.e., in their ionization chambers.

Let us recall that the ionization of a neutral atom consists either in the detachment of an electron or, conversely, in the addition of an electron. At the present time, only the first ionization process is utilized in EPE. The energy required for the detachment of an electron is characterized by an "ionization potential" α_i, measured in volts. The energy itself necessary for the ionization of a single atom is equal to the elementary charge multiplied by the ionization potential. The ionization potentials of substances most frequently used in EPE are as follows: Li, 5.4 V; Cs, 3.9 V; Hg, 10.5 V; Xe, 12 V. /24

Two ionization mechanisms are of interest for modern EPE. The first involves ionization by electron impact, when an electron of sufficient energy strikes an atom and knocks another electron out of it. However, electrons do not necessarily ionize an atom on striking it. They may be reflected elastically or merely excite the atom. As a result, the actual energy expended in the formation of an ion appreciable exceeds the ionization potential, and in perfected systems usually amount to about 50-100 eV per ion. The energy required to produce a single ion is called the ion rating.[1]

The second ionization method involves ionization of atoms in contact with a solid. It takes place when the electronic work function W (i. e., the energy of detachment of an electron from the surface of the emitter) is greater than the ionization work (energy) of a given atom. For tungsten, W = 4.5 eV, and for cesium, the ionization work is 3.9 eV. Therefore, tungsten effectively "sucks in" an electron of the cesium atom, converting it to an ion. The ion formed remains on the surface of tungsten for a certain time, until the surface, which "vibrates" as a result of thermal motion, sheds it.

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1. Thus, the generation of an ion current of 1 A for a rating of 100 eV/ion requires a power of 100 W.

It follows directly from this picture of surface ionization that the emitter (in this case, tungsten) must be heated. Otherwise it would be covered by a layer of adhering cesium ions, the work function of the surface would markedly decrease, and its ionization capacity would disappear. It turns out that for the tungsten-cesium pair, the working temperature of the surface should be about 1500°K. In connection with the foregoing, ion engines are divided into two groups: those with volume ionization (VIE) and those with contact ionization (CIE).

Ion engines with volume ionization. The source of ions in engines with volume ionization is a plasma produced by a discharge between the anode and cathode in a special gas-discharge (ionization) chamber. This type of EPE is therefore called a plasma-ion engine.

We shall examine a typical plasma-ion engine of G. Kaufman's design (Fig. 8). Clusters of these engines have been proposed for use in cruise missions[1] in the American space program SEP, which will be discussed in the next section. The ionization chamber of the engine (emitter) is in the shape of a cylinder, on one end of which is mounted a hollow cathode; the cylinder walls are the anode, and the opposite end, adjacent to the accelerating gap, is made in the /25

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1. The missions entrusted to jet engines in space may be divided into four categories: correction of SC orbit, altitude of SC in space, stabilization of SC orbit and cruising maneuvers - insertion of SC in a given orbit and transfer of SC from one orbit to another.

form of a grid through the openings of which the plasma ions are "pulled out" by the electric field into the accelerating gap. A longitudinal magnetic field (more accurately, in the shape of a gramophone tube) is induced by magnets inside the

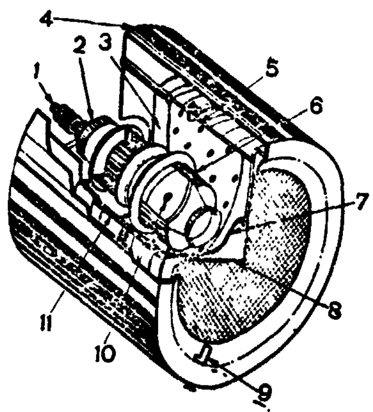

Fig. 8. Diagram of plasma-ion engine: 1 - evaporator; 2 - insulator; 3 - mercury vapor supply; 4 - electrostatic shield; 5 - anode; 6 - pole of magnet; 7 - grid of discharge chamber; 8 - accelerating electrode; 9 - neutralizer; 10 - cathode; 11 - permanent magnet.

chamber. The emitter grid and accelerating electrode are made in a convex shape to avoid a local change in the width of the accelerating gap due to thermal deformations of the structure. The outer electrode (screen) is the engine casing, on which the neutralizer is mounted. The engine and neutralizer operate on mercury stored in special tanks (separate for the engine and neutralizer). The mercury, which enters the engine through a tube, passes through the evaporator insert made of /26 porous tungsten, and then the mercury vapor penetrates through a porous insulator and enters the high voltage part of the engine. A distributor delivers part of the vapor (about 10% of the total) to a hollow cathode, and the remaining vapor is introduced through openings in the end, on which the cathode is mounted, directly into the chamber. When the discharge voltage is applied across the cathode and anode. a discharge is fired. The electrons formed strike the neutral atoms of the working emdium and ionize them. A plasma is thus formed. The magnetic field is produced in order to reduce the loss of ions and electrons due to their deposition on the chamber walls. The hollow cathode - a neutralizer inside which a discharge is taking place - supplies the plasma. The plasma electrons travel along the plasma "bridge," enter the emerging beam of accelerated ions, and neutralize it.

The diameter of the ionization chamber of the thruster is 30 cm. The thruster weighs a total of 7 kg and consumes 2.75 kW of power. The current intensity is 2 A. Under rated conditions, the engine (emitter potential + 1kV, accelerating electrode potential 0.5 kV) develops 13.5 g of thrust at a specific impulse of 3000 sec and a thrust efficiency of 73% (thrust value, 200 W/g; ion rating, 250 eV/ion). The engine can also operate at a power level 3/4 and 1/2 of rating at thrust efficiencies of 71.5% and 67% and specific impulses of 2970 and 2810 sec respectively. The calculated service life (total operating time) of the engine is 10,000 h. In the future, it is planned to increase the service life to 20,000 h with a certain increase of the specific impulse (to 4000-5000 sec), and as the working medium, to use xenon, which is much more convenient than mercury, both in storage and when used aboard SC. Preliminary tests have shown that the characteristics of the engine in xenon operation remain almost unchanged.

Ion engines with contact ionization. From the standpoint of the production of ions for the operation of such an engine, a certain combination of the working medium with the material of the emitter is required (see above). The pair, cesium (working medium) - tungsten (emitter material) is used most frequently. Structurally, the ion emitter is made in the form of a plate of porous tungsten; cesium vapor is passed through the pores. The best results are obtained by using emitters with an ordered structure, prepared by /27 sintering tungsten powder consisting of globules of definite diameter (on the order of a micron). The coefficient of transformation of cesium atoms into atoms when such emitters are used is very high, close to 100%. A major advantage of an engine with contact ionization is the absence of a bulky ionization chamber with a rather "capricious" discharge. In addition, the surface emitting the ions is rigid, not movable, as in a plasma-ion engine. This markedly improves the engine control characteristics.

However, the porous emitter has significant drawbacks. In the first place, even at the working temperature of the emitter, of the order of 1500°K, which is comparatively low for tungsten, a further sintering of the emitter and closing up of the pores takes place in the course of the operation, and therefore the service life of an ion engine with contact ionization proves more limited than that of a plasma-ion engine. In the second place, the heating of the fairly massive and intensely radiating emitter consumes considerable energy, so that the ion rating increases (to 1000-3000 eV/ion), and the energy efficiency declines. In contrast to the advantages, these drawbacks appear to be of a temporary nature.

Today, ion engines with contact ionization are of inter- only in the microengine version, when the absolute energy losses due to heating of the emitter are small. An example is the American engine made by the EOS Co., tested is space on the ATS-4 satellite. Cesium from a tank flows along a porous nickel rod into an evaporator, and thence into a single-cell emitter. The annular copper accelerating electrode is divided into four sections, so that by supplying an additional voltage to two opposite sections, one can deflect the ion beam by approximately 10° in two mutually perpendicular directions. Two thermionic neutralizers, main end and backup, are attached to the outer electrode. The electron source in the neutralizer is an incandescent tantalum wire with a 5% admixture of yttrium. The emitter potential is +3000 V, and the potential of the accelerating electrode is -2000 V, the ion beam current being about 1 mA. The engine with the power supply and voltage conversion systems weighs less than 2.7 kg. It is designed for four rated operating conditions. The /28 parameters of one of them are: thrust, 9 mg; power consumed, 26 W; specific impulse, 6700 sec; thrust efficiency, 14.6%. The power consumed by the emitter is 10 W, and the neutralizer consumes 3.5 W. The conversion coefficient of the working medium is close to 100%.

Summarizing the above, we note that the great effort which has been expended in the design, construction and optimization of plasma ion engines has been crowned with success: modern plasma ion engines have high characteristics and operate with great stability and reliability. However, the creation of contact ion engines of high characteristics remains to be accomplished in the future.

Plasma engines. A plasma is a mixture of neutral atoms, ions and electrons. The charge concentration[1] of the ions and electrons in a plasma are equal with a high degree of accuracy. A plasma is therefore said to be quasi-neutral. Owing to the presence of moving charged particles, a plasma is a good conductor of electric current.

Plasma engines combine the features of electrically heated (gas-dynamic) and ion engines. They are related to the former by the quasi-neutrality of the accelerated medium, and to the latter, by the use of electromagnetic forces for the acceleration.

A plasma can be accelerated by both the thermal energy of the plasma (i. e., by a pressure drop) and the ampere force. We shall accordingly distinguish electrothermal and "electromagnetic" plasma engines. There is not distinct dividing line between these categories, and many specific engines (for example, end-type and pulsed plasma engines) can switch from thermal to electromagnetic operating conditions when the power level is changed.

Interpretations of the two indicated acceleration mechanisms are useful, as a rule, in cases where the plasma density

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1. That is to say, products of the concentration of particles of a certain kind and their charge, expressed in units of elementary charge.

is high, or, more accurately, when the mean free path of the plasma particles between collisions is short in comparison with the characteristic dimensions of the acceleration zone. If this condition is not fulfilled, it usually more conven- /29
ient to consider the motion of the electron and ion components of the plasma separately. In view of the enormous difference in the ion and electron masses, plasma acceleration should be taken to mean the acceleration of ions with preservation of quasi-neutrality.

In the accelerating channel, the plasma ions and electrons are acted upon by electric and magnetic fields, and also by collisions of particles with one another. Selecting some "typical" singly charged ion, we can write the equation of motion (Newton's second law) for it in the following form: $Ma_i = eE + (e/c)\ [v_i \times B] + F_{ii} + F_{ie}$.

Here M is the mass of the ion, e is its charge, v_i is its velocity, a_i is its acceleration, E and B are th intensities of the electric and magnetic field respectively, F_{ii} is the force produced by collisions of ions with one another, and F_{ie} is the force produced by collisions of ions with electrons. A similar equation can be written for a "typical" electron of mass M. Assuming that the electronic charge is negative and that the momentum of the electron-ion system is preserved during the collision of an electron with an ion, we have $ma_e = -eE - (e/c)\ [V_e \times B] + F_{ee} - F_{ie}$.

It is obvious from the above equations that the acceleration of ions (i. e., plasma) may occur as a result of three factors: electric field, collisions with ions, and collisions with electrons. The magnetic field acts on the ion with a force perpendicular to the ion velocity (Lorentz force), and therefore does not change its energy.

The mechanism of acceleration of ions by an electric field is identical to that operating in ion engines. However, now the acceleration channel (gap) contains not only ions, but also electrons, whose concentrations may be considered equal with a high degree of accuracy, since, on the average, the plasma is electrically neutral. It is easy to see that if the particles are acted upon only by an electric field, the acceleration of a plasma as a system of electric-ion pairs is absent: the force acting on an electron is numerically equal to that acting on an ion, but these forces are acting in opposite directions, so that the total impulse of the electron-ion pair and hence, impulse of the plasma remains unchanged. How the electric field can be forced to "work" will become apparent a little later.

The second acceleration mechanism involves ion-ion collisions. Such collisions are capable of converting the thermal energy of ions to energy of directed motion. As applied to atoms and molecules, this mechanism operates in ordinary rocket engines and electrically heated EPE. For EPE with /30 high exhaust velocities, this cannot be the basic mechanism, since it would require a very high plasma temperature. However, it plays an important service role in many cases.

Finally, ions can be carried off by a stream of electrons ("electron wind") as a result of collisions. This mechanism begins to play a particularly important role in cases where the velocity of relative motion approaches the thermal velocity of electrons, since this is associated with the buildup of intensive oscillations in the plasma that sharply increase the effective interaction of the electrons and ions. The mechanism of acceleration by the "electron wind" is decisive in so-called end-type engines.

To describe the plasma dynamics as a whole, we shall formulate the equations of motion of its components. We thus obtain $Ma_i + ma_e = F_{ii} + F_{ee} + (e/c)\ [(v_i - v_e) \times B]$. The first two terms on the right are responsible for the thermal acceleration of the plasma, and the third term is equivalent to the current force - this is evident even from the fact that it includes the velocity of relative motion (i. e., the current velocity) and the magnetic field.

We shall now consider specific types of plasma engines.

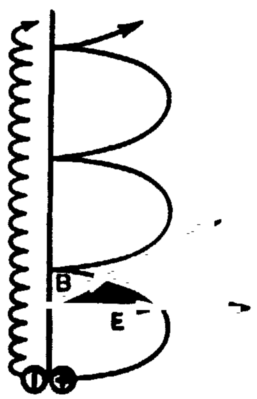

Fig. 9. Trajectories of charged particles in mutually perpendicular electric and magnetic fields.

Plasma engines with azimuthal drift. In these electromagnetic engines, the ions are accelarated by an electric field. To explain the principle of their operation, we shall consider the motion of a single charged particle in mutually perpendicular electric and magnetic fields (Fig. 9). If the particle was originally at rest, it begins to accelerate under the effect of the electric field. This gives rise to a Lorentz force, perpendicular to the velocity of the particle and to the magnetic field, a force that turns the particle /31 around and a a certain instant forces it to go against the electric force. The particle slows down, stops, and the process is repeated over again. In the nonrelativistic case, the motion of the particle takes place along a cyloid and is composed of displacement in a direction perpendicular to both the field E and the field B, at a drift velocity[1] u_{dr} (km/sec) = 10^3 E (V/cm)/B (G) and rotation in the magnetic field at the "cyclotron" frequency. The radius of this rotation ("Larmor" radius) for electrons is much smaller than for ions, in proportion to the difference between the electron mass and the ion mass. This helps to force the electric field to effect the acceleration.

We shall take a coaxial (tubular) channel with a radial magnetic field and longitudinal electric field, and choose its length so that it is large in comparison with the elec-

1. In the CGSE system, u_{dr} = cE/B, where c is the velocity of light.

tronic Larmor radius, but small in comparison with the ionic Larmor radius. If a plasma is now produced in the channel, the ions, accelerated by the electric field, will be essentially undeviated by the magnetic field, and will fly through the accelerating gap, whereas the electrons will drift along the azimuth (i. e., around the axis of the system), without being able to leave the accelerating gap. The energy of accelerated ions is determined by the potential difference which they traverse. At the exit, the ion beam must be neutralized by electrons from the compensator cathode (neutralizer). The plasma density is chosen so that the collisions of the particles with one another in the acceleration zone are rare. We have thus obtained an acceleration channel for a plasma engine with a closed drift, in which the ions acquire a velocity under the influence of the electric field in the presence of electrons compensating for the space charge of the ions.

All we have to do is provide for a continuous supply of ions to the channel. This takes place as follows: the atoms of the working medium coming from the anode are ionized near the anode in a rotating electron cloud, the ions obtained are then accelerated, and the electrons "generated" together with them as a result of collisions and oscillations /32 strike the anode and, having "passed around" the electric circuit, escape from the compensator cathode and leave the system together with the ions.

Making a comparison between the above-described engine with azimuthal drift and an ion engine, we see that they resemble each other in a certain sense: in both systems, acceleration of ions takes place in an electric field, followed by neutralization of the ion beam. Actually, the properties of these systems are, on the whole, sharply different.

First of all, the accelerating gap of an engine with azimuthal drift contains a quasi-neutral plasma, and thus, the space charge does not impose a limitation on the ion current.[1] For this reason, such an engine can operate at the most diverse accelerating voltages, including small ones (on the order of 100 V), this being practically unattainable for ion engines because of the low ion current extracted in this case. The ion current in good models of engines with azimuthal drift is very close to the discharge current[2] and is determined solely by the magnitude of the mass flow rate of the working medium. Thus, in engines with azimuthal drift, in contrast to ion engines, it is possible to change the mass flow rate and accelerating voltage independently, i. e., the thrust and exhaust velocity, over a wide range (by a factor of 3-5), while maintaining a high

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1. From an electrodynamic standpoint, the plasma is accelerated by the ampere force of interaction of the radial magnetic field with the azimuthal electric (drift) current.

2. In the ideal scheme discussed above, the discharge current is simply equal to the current of the ions ejected from the engine.

efficiency. The engine with azimuthal drift thereby affords a real possibility of carrying our optimum maneuvers of the SC.

In addition, instead of the precision electrodes of ion engines with numerous holes, the function of ionic "optics" in engines with azimuthal drift is played by the magnetic lines of force, since they keep the electrons from moving in the direction of the electric field. Because of the high mobility of electrons, the magnetic lines of force in the plasma are electrically equipotential. i. e., the electric field is always orthogonal to them. This makes it possible to control the focusing properties of the system by creating /33
the necessary magnetic field configurations.

Two modifications of engines with azimuthal drift exist: with an extended acceleration zone and with a narrow acceleration zone. They differ in the ratio of the length of the accelerating gap to the electronic Larmor radius.

The engine with an extended acceleration zone[1] (Fig. 10) operates on xenon, which enters the channel through holes in the anode. The length of the acceleration channel is large in comparison with the electronic Larmor radius (it is approximately several tens of times greater than this radius), and the electric field in the plasma is relatively small, on the order of 50 V/cm. The walls of the channel are dielectric. A magnetic field (basically radial) with

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1. It is also called the stationary plasma engine.

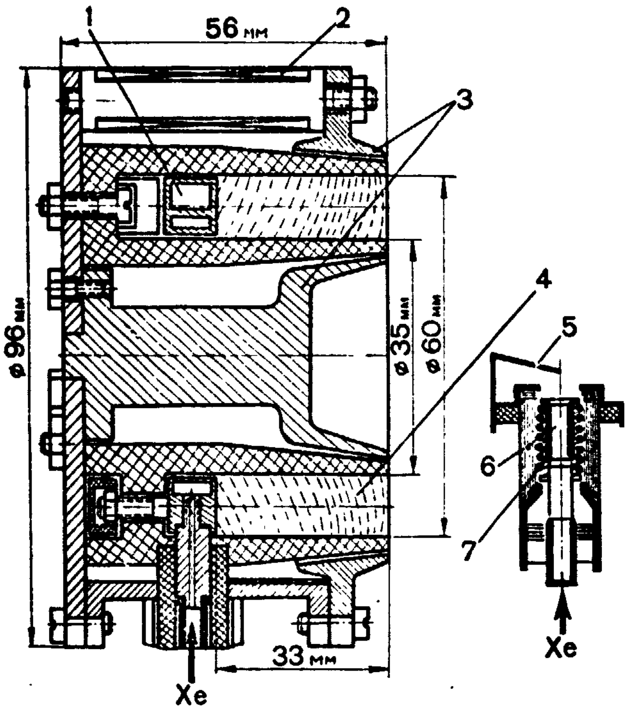

Fig. 10. Diagram of engine with azimuthal drift and an extended acceleration zone: 1 - anode; 2 - coils (8 units); 3 - poles; 4 - acceleration channel; 5 - ignition electrode; 6 - tube of lanthanum hexaboride; 7 - starting spiral of heater.

an intensity of about 200 G is induced by a system of solenoids and a magnetic circuit. This engine was tested in space on the "Meteor" satellite. Under rated operating conditions, it consumes a power of 400 W, producing an ion current of 2.5 A at an accelerating voltage of 160 V, and develops a thrust of about 2 g with a specific impulse of 1000 sec. By varying the flow rate of the working medium and the power supplied, one can obtain a thrust from 1 to 8 g with a specific impulse between 1000 and 2500 sec with this engine. The engine weight is 1.5 kg. /34

In addition to the advantages of the engine with an extended acceleration zone enumerated above, one should note two or more advantages. First, this engine is highly reliable and stable in operation, and second, it has an exceptionally low thrust rating, less than 200 W/g, being second in this respect only to electrically heated EPE.

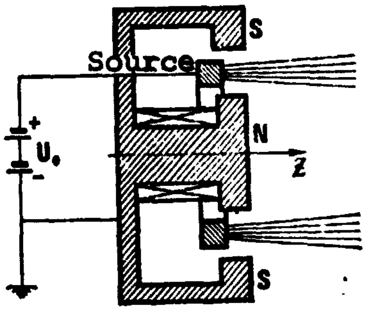

Fig. 11. Diagram of engine with azimuthal drift and a narrow acceleration zone.

In an engine with a narrow acceleration zone[1] (Fig. 11), the rotating electron cloud has a thickness of the order of only a few electronic Larmor radii. The channel walls are metallic; the applied potential difference is concentrated in the layer near the positive electrode. Two stages are usually constructed in modern models: ionization of atoms on the working medium takes place in the first, and the ions obtained are accelerated in the second. These are high-voltage systems with high electromagnetic fields. At a voltage of several kilovolts in a magnetic field on the order of 1 kilogauss, these engines generate a beam of bismuth ions with a current of 10 A and a very high energy efficiency (from 70 to 80%). The ion beam obtained is sufficiently monoenergetic. /35

Engines with azimuthal drift are structurally simple and capable of operating reliably and stably over a wide range of powers and thrusts with different specific impulses. They unquestionable deserve the closest attention, further study, and improvement.

<u>Plasma pulsed engines</u>. These engines are, so to speak,

the "senior" ones in the large family of EPE. Essentially, the engine built by V. P. Glushko and tested in the 1930's was a prototype of modern pulsed plasma engines with all their basic attributes. The above diagram (see Fig. 5) of acceleration of a connector under the influence of the ampere force in an intrinsic magnetic field represents the design of a pulsed engine.

In studying the discharge in gaseous pulsed plasma engines, scientists have encountered a curious phenomenon: as the gas supply decreased, the parameters of the system ceased to depend on the gas flow rate. It was thought that the discharge itself "chooses" the amount of substance it needs from the electrodes. In designing pulsed plasma engines, this fact suggested the idea of deliberately "supplying" to the discharge an easily vaporizable solid dielectric (introduced into the inter-electrode gap) from which the charge would "bite off" the portions it needed. This made it unnecessary to inject gas with complex pulse valves. At the present time, only pulsed plasma engines operating on solid dielectrics - the so-called "erosion" pulsed plasma engines - are used for space missions. As a rule, teflon is used as the dielectric.

As an example, we shall examine a pulsed engine designed by the Fairchild - Hiller Co. and tested on the American satellite LES-6. It operates on teflon. Its appearance and diagram are shown in Fig. 12. A teflon rod acted upon by the /36 pressure of a spring is pushed into the interelectrode gap, where it is pressed against a projection (rib) made on the anode. Thus, the working surface of teflon is always accurately fixed. An "ignition" system, which initiates the charge that evaporates and ionizes a certain small amount of dielectric, is mounted on the catnode. The firing precedes the main discharge. The latter, which vaporizes, ionizes and accelerates the bulk of the dielectric, is produced with a 2 μF capacitor charged to a voltage of 1360 V (energy, 1.85 J). Each discharge carries off 10^{-8} kg of teflon (about 8×10^{16} molecules). For greater reliability of operation,

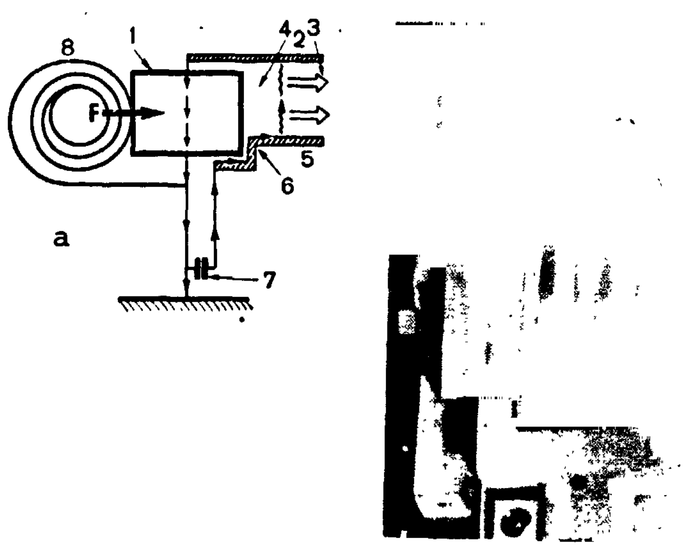

Fig. 12. Erosion pulsed plasma engine: a - diagram: 1 - telfon rod; 2 - cathode; 3 - plasma jet; 4 - ignition devise; 5 - anode; 6 - ridge; 7 - capacitor; 8 - feed spring; b - appearance of engine

two subsystems of anodes, cathodes, ignition devices and teflon rods are mounted in the engine with one discharge capacitor, without causing any appreciable increase in engine weight. The specific impulse of the engine is 310 sec, although ions with a velocity of 40 km/sec are observed. This attests to the low ionization of teflon (the degree of ionization is only 7.5%). The engine efficiency is also low, 1.8%. The duration of the discharge pulse is slightly below 3 μ sec. The engine with its casing weighs 1.3 kg (each teflon rod weighs 100 g) and is designed for 12 million discharges with an impulse of 2 mg·sec each. /37

The low efficiency of pulsed microengines must not be ragarded as a fundamental flaw of pulse plasma engines. This is a characteristic property of all microengines, and is explained mainly by the impossibility of reducing the unproductive energy losses to zero. Since they always remain infinite, these losses lead to low efficiency values when the energy consumed by the engine is low. With increasing energy expended in the discharge, the efficiency of pulsed engines increases.

There is no question that for a whole series of problems, the pulsed mode of engine operation is not as good as the stationary mode. However, if an attempt is made to determine the importance of pulsed plasma engines among EPE, one must mention their significant inherent advantages, which make these engines fully competitive with other EPE. In the first place, the construction of pulsed plasma engines is extremely simple. In the second place, their energy and thrust range is perhaps the broadest of all the existing

EPE, since by varying the pulse frequency, one can change the time-averaged thrust at will. In the third place, they can operate with the most diverse working media; even a simple pebble, which would make any other modern EPE "run", will be "edible" to a pulsed plasma engine. Finally, in the fourth place, they are extremely reliable in operation. All this suggests that in the future, the use of pulsed plasma engines will be very popular.

End-type plasma engines. These engines have resulted from the evolution of pulsed coaxial plasma accelerators with an intrinsic magnetic field. The evolution began with the creation of a stationary coaxial plasma accelerator which differed from a pulsed accelerator in that the working substance was continuously supplied to the interelectrode gap, and the electrical supply was steady. It was later found that extended coaxial electrodes led to a com- /38
plex pattern of plasma flow near the electrodes, as a result of which the plasma was pressed against the central electrode, and the efficiency of the accelerator decreased sharply. For this reason, the inner electrode began to be shortened and displaced into the interior of the discharge chamber. Systems of such configuration were named end-type accelerators. They were used to construct end-type plasma engines (they are also called plasma-arc or magnetoplasmadynamic engines).

In end-type plasma engines, between the anode and the cathode, there is generated an arc discharge which ionizes and heats the working substance, supplied most frequently through the inner cathode electrode. The electrodynamic ampere force produced by the interaction of the electric field with the intrinsic magnetic field accelerates the plasma along the axis of the system and at the same time compresses the plasma jet in the radial direction. In the axial region, the plasma acceleration is of thermal origin, since the ampere force is small there[1]. End-type plasma engines with an intrinsic magnetic field are powerful, high-current systems : the discharge current under steady conditions reaches 10 kA for a consumed power of hundreds of kilowatts. On the contrary, their flow rate level of the working substance is low (on the order of hundreds of milligrams per second). The low flow rates distinguish end-type plasma engines from plasmatrons, which are widely used in engineering (in the latter, the characteristic flow rate is measured in grams per second). As a result, the electrodynamic force in end-type plasma engines turns out to be applied to a small mass, so that the exhaust velocity obtained is high. Easily ionizable alkali metals, for exemple lithium or potassium, are most frequently used as working media in end-type plasma engines.

1 Of the elementary acceleration mechanisms, the following two operate in end-type plasma engines: ion-ion collisions and "electron wind".

As an example of an end-type high-current engine with an intrinsic magnetic field, we can take an engine operating on liquid bismuth, supplied to the inner conical surface of the cathode in the form of a thin film on which the discharge takes place (fig. 13). The discharge current in the engine changed from 3 to 8 kA, and the power consumed, from 175 to 800 kW; the flow rate of the working substance is 0.5 to 1g/sec. An exhaust velocity of about 20 km/sec was reached at 30% efficiency. The engine thrust ranged from 0.5 to 2.5 kg and was proportional to the square of the discharge current. /39

At a low discharge current and a level of consumed power of several kilowatts (up to approximately 20 kW), the operation of an end-type high-current engine is inefficient: the intrinsic magnetic field is unable to provide for an adequate compression of the plasma jet and an effective electrodynamic acceleration. Under these conditions, good results are obtained by applying the external magnetic field with solenoids attached to the outside of the engine. Such engines are called end-type Hall engines; in the latter, the accelerating force is generated by the interaction of an azimuthal Hall electric current (similar to the current in a engine with azimuthal drift) with the radial component of the external magnetic field. The thrust of an end-type Hall engine increases linearly with the discharge current and external magnetic field. Many-hour tests

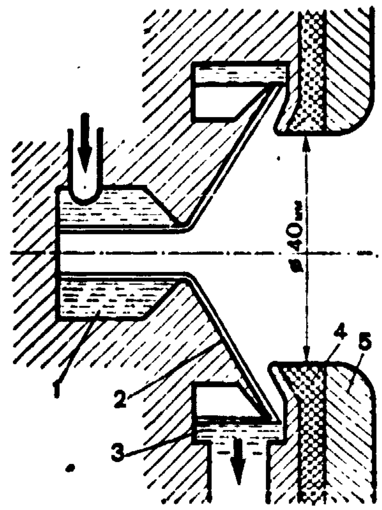

Figure 13. Diagram of
end type high current engine:
1- liquid bismuth; 2- cathode;
3- drain; 4-neutral insert;
5- anode.

of such engines with lithium confirmed that it is possible to obtain specific impulses of 4000-5000 sec at 50-60% engine efficiency and 25-40 kW consumed power. At lower powers (a few kilowatts), a specific impulse of 2000 sec with 35 % engine efficiency was obtained. /40

An effort to operate with a high-discharge current at low flow rates of the working substance, dictated by a desire to produce an effective electromagnetic acceleration of the plasma, has met an unexpected obstacle. While the cathodes of end-type engines can operate steadily at a very high current density (up to k 1 kA per square centimeter of cathode surface), the situation is much less hopeful with anodes. At high discharge currents, a region of a strong electric field (anode potential jump) is formed near the anode, and on the anode itself, and current concentrations [1] which destroy it are formed. It was also found that at a given flow rate of the working substance, after having reached a certain value, the discharge current almost ceases to increase with rising consumed power, but on the other hand, the anodic potential jump and the anodic erosion increase sharply, and marked oscillations of the discharge parameters appear. This phenomenon, which was named "current crisis," characterizes all end-type engines. It is not yet quite clear precisely how the "current crisis" should be overcome or whether it should be used for plasma acceleration.

The currently available on-board space power equipment is unable to provide a normal supply for engines as powerful as end-type ones. For this reason, these engines, designed primarly for cruising missions, have not yet been tested in space.

Modes of development of EPE. We shall now try to visualize the modes of further development of EPE. Improvement of our knowledge of the operating processes in the types of EPE described, and, so to speak, an all-inclusive improvement of the elements of these EPE, as well as an extension of their working ranges, will constitute the immediate objectives of the near future.

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1 Concentration of a significant fraction of the discharge current at one or several points (or regions of small area) of the anode surface.

Principal attention will be directed, not so much to increasing the thrust characteristics, since they are in general already sufficiently high, as to the improvement of the operating characteristics, primarily increasing the service life and reliability. However, it is perfectly obvious that along with the operation of the indicated character, there will /41 take place a search for qualitatively new engines along the lines of both merging of already existing designs and creating engines based on qualitatively different principles.

What dictates the necessity of this process? It is due to different causes, but in the final analysis, to the problem of maximum adaptation to the "natural" conditions in the spacecraft, similar to the case of life forms in the course of evolutionary development becoming more perfectly fitted into their ecological "niche". In this respect, plasma engines are particularly promising, since the plasma has an enormous number of internal degrees of freedom. It is for this reason that so much is being said about the instability of a plasma, its "capriciousness." However, it goes without saying that a time will come when each degree of freedom will be put under control.

We shall discuss in somewhat greater detail the problem of matching EPE with sources of power supply and the problem of reducing unproductive losses due to ionization of the working medium. We shall introduce two parameters. One is called the exchange parameter and may be defined as the ratio of the mean kinetic energy of the ion W_i to the discharge voltage U_d : $\xi = W_i/eU_d$, provided that the engine efficiency is 100%. The second parameter is the ionization coefficient α . It is the ratio of the number of ionized atoms to the total number of atoms entering the channel, provided that both the ionized and the unionized atoms participate equally in the process. At first glance, it may be thought that all EPE operate only when $\xi = 1$ and $\alpha = 1$. This is not so. Figure 14 shows in ξ α coordinates the regic. s of operation of different engines with 100% efficiency. We see that the point $\xi = 1$, $\alpha = 1$ is indeed "overloaded." It corresponds to ion engines and engines with azimuthal drift. Naturally, electrically heated EPE lie on the line $\alpha = 0$. In principle, ionization is not required for them. However, end-type engines lie on the straight line $\alpha = 1$ /42 primarily when $\xi > 1$. This fact, which may seem strange at first glance, is explained by the fact that, as already mentioned, the ions in an end-type engine are accelerated not by an electric field, but by an "electron wind," a ıd here the stable relationship between W_i and eU_d disappears. Attention is drawn to the region of colloidal engines. These engines are notable by the fact that their ionization losses are very low. Finally, pulsed engines can operate over a wide range of degrees of ionization, since they involve the acceleration of a dense plasma, in which the ions and neutral atoms quickly "equalize" their velocities as a result of collisions. There

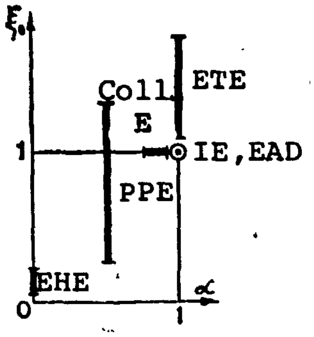

Figure 14. Diagram of (ξ, α) EHE - electrically heated EPE; PPE - pulsed plasma EPE; IE- ion engines; EAD - engine with azimuthal drift; ETE - end-type engines; coll. E - colloidal engines

is no doubt that the search for new EPE will be aimed at finding engines that function at those ξ and α that are the most "natural" for certain conditions of a given of a given SC.

EPE Operate in Space

What precedes space tests. Laboratory studies of EPE have led to the creation of fairly reliable and effective engine models in 1963-65. It was necessary to check their operation experimentally under space conditions. However, the transition from laboratory models, which were essentially physical instruments, to experimental space engine units was far from simple. It was necessary to create a space version of a closed automated system controlled from earth by a small number of commands.

Thus, scientists and engineers were confronted with a new series of complex problems. It was necessary to build reliable /43 and efficient converters of electrical energy with on-board parameters into energy with the parameters that were necessary for the operation of EPE. It was necessary to develop a system of storage of the working medium and systems for supplying it and adjusting its flow rate. The unusually low flow rates (milligrams per second instead of the usual grams and kilograms) and long service lives of the engines required a completely new approach.

It was also necessary to reduce the weight of both the engines themselves and the propulsion units as a whole, to give them the functionally most effective shape, to observe the rigid dimensional requirements imposed on space electric propulsion units of SC. All this had to be "connected" by monitoring, control and telemetry systems.

There is one more important fact. An experiment is only an experiment. This being the case, a spacecraft on which an electric propulsion unit is mounted accumulates various measuring systems which monitor the electrical parameters of the engine, measure the thrust, the coating of the SC surface with the working medium escaping from the EPE, the electric potential of the craft, etc. Earthbound services become involved in the space experiments. They must study the transmission of commands during the operation of EPE, measure the parameters of the plasma "tails" formed behind the EPE, simulate on stands various malfunctions that may occur in space, etc.

Two basic questions predominate over all these considerations: will the equipment prepared with such great care fail in space, and will its parameters correspond to the results of a laboratory tests?

Modern EPE, including those developed most recently, should have a service life measured in thousands or tens of thousands of hours under conditions where repairs and adjustments are impossible. The length of trouble free operation of EPE can be estimated from the fact that the service life of an ordinary electric bulb is only 1000 h.

The answer to the question of whether the "space" parameters of an engine unit will correspond to the laboratory parameters is much less clear. This is because the conditions of /44
operation in space differ considerably from those created on laboratory test benches. These differences pertain to the most diverse aspects, but they are primarily related to the fact that the dimensions of the vacuum chambers in which EPE are tested are fundamentally limited. In many cases, this may radically affect the operation of EPE. In addition, the best vaccum obtained in these chambers can in no way compare with the vacuum of space, and the influence of the residual gas on the operation and parameters of EPE is also impossible to estimate with absolute accuracy.

As an example of the problems that can be reliably solved only in a space experiment, we should mention the following: the problem of neutralization of ion fluxes ejected from ion engine and engines with azimuthal drift; the problem of coating of the engine parts and SC with erosion products of EPE walls and with the working medium ejected from the EPE, and the level and direction of radio noise.

The first EPE successfully tested in space were ion and pulsed erosion plasma engines (1964). Electrically heated engines were texted next. In 1971, prolonged space tests of ion engines were conducted. They included tests of stationary plasma engines with azimuthal drift and an extended acceleration zone. We shall examine the principal results of these space experiments.

"Zond-2" in space. The first successful tests of EPE on a satellite were carried out in the Soviet Union. In early December 1964, the automatic interplanetary station "Zond-2" was launched in the direction of Mars, with six erosion-thermal pulsed plasma engines aboard. The testing program included the maintaining of a constant solar orientation of panels of solar batteries which, when deployed, were to charge storage batteries. The engines were also powered by solar batteries.

Fifteen days after launching, at a distance of over 5 million km from earth, on radio command, the attitude system of the "Zond-2" station was switched to pulsed plasma engines. The space electric propulsion unit worked for the specified 70 min, steadily maintaining the required orientation of the solar batteries. Thus began the operation in space of pulsed electric propulsion engines.

On 26 Septembre 1968, the American communications satellite LES-6 was launched, on which was mounted an electric jet propulsion unit which included four Fairchild-Hiller engines described above, and an energy conversion system that weighed 0.9 kg. The satellite was spin-stabilized, and the engine unit maintained it in a given orbit. The unit operated in the single-pulse mode for 1.5 sec; the engines were activated on 15 October 1968. The engines operated normally in space for over two years.

Short tests of ion engines. The purpose of the initial tests of ion engines was a qualitative explanation of their operation and neutralization of the ion beam under space conditions.

In january 1964, aboard the rocket "Blue Scout Junior," which completed a flight along a ballistic trajectory, a cesium ion engine with contact ionization was activated, which functioned for a slightly shorter period than specified.

In July 1964, under the SERT-1 program, two ion engines, one made by Hughes Aircraft and one by the Lewis Research Center, were tested on the Scout rocket (50 min of ballistic flight). The mercury engine of the Lewis Center generated a specific impulse of 6000 sec, 2.3 g of thrust, and weighed 5.3 kg. It functioned normally for half an hour, during which it was turned on and off repeatedly. The Hughes Aircraft engine did not function.

In April 1965, the Snapshot satellite with a cesium contact ion engine was placed in orbit. The engine operated normally for about an hour. Thus, the feasibility of operation of an ion engine and effective neutralization of the ion beam in space were qualitatively confirmed.

The first detailed studies of the operation of plasma-ion engines with thermionic and plasma neutralizers were conducted in the Soviet Union under the "Yantar" program and were initiated in October 1966. The "Yantar" automatic ionospheric /46
laboratories were lifted to a height of 400 km by geophysical rockets. The laboratories were launched into space four times along ballistic trajectories. The electric field, density of the ion current from the ionosphere and electric potential on the surface of the laboratory, as well as the discharge current of the engine and ion beam current were measured during the tests.

The argon plasma-ion engine was tested first. The engine was turned on at a height of 160 km; 11 on-cycles were carried out. The specific impulse recorded was found to be 4000 sec. During subsequent flights of the "Yantar" laboratories, nitrogen and air-plasma ion engines were tested; they both functioned reliably and stably. The nitrogen operation developed a specific impulse of slightly more than 12,000 sec, and the air operation developed 14,000 sec.

Tests under the "Yantar" program showed the presence of effective neutralization of the ion beam by electrons from the plasma neutralizer (the potential of the laboratory housing did not exceed 0.3 % of the accelerating voltage). The operation of the thermionic neutralizer was somewhat poorer.

On 10 August 1968, the American satellite ATS-4, which carried two cesium contact microengines of the EOS Co., described above, was placed in orbit. The tests confirmed the workability of contact ion engines and efficiency of neutralization of the ion beam. They were followed by prolonged tests of ion engines.

Tests of ion engines under the SERT-2 program. The SERT-2 program continued SERT-1. The tests were designed to meet the following objectives: confirm the reliability and determine the service life of ion engines, demonstrate the compatibility of space electric propulsion units with SC systems, design and refine service operations, determine the operating characteristics of the engine units in space, and confirm the results of earthbound tests.

The SERT-2 spacecraft with two ion engines on board was launched on 3 February 1970. A sun-synchronous SC orbit 1000 km high was chosen.[1] On the "Agena" rocket stage were mounted two /47 panels of solar batteries providing an electric power of about 1500 W, as well as a block of service systems (systems of

1 So that its plane precessed at an angular velocity of 1° a day, which is the angular velocity of rotation of the earth around the sun. This was done in order to provide for a continuous illumination by sunlight of the panel with the photocells (solar batteries), which were the main source of power for the EPE.

energy conversion, switches, telemetry and command systems, attitude control) and the SC itself with the ion engine unit and experimental equipment (probes for measuring the SC potential, device for determining the effect of the jet on the solar photocells, and instrument for estimating the effect of the operation of the EPE on radio communication, an accelerometer for measuring the thrust, etc).

Two space electric propulsion units were mounted on the SC, each of which included a Kaufman ion engine and an energy conversion system. The engine, with an ionization chamber 15 cm in diameter, ran on mercury, and, while consuming a power of 860 W, generated a beam of accelerated ions with a current of 0.25 A and a specific impulse of 4240 sec. The engine efficiency was 68 %. It developed 2.8g of thrust. The mercury reserve (14.5kg), designed for 5800 h of operation, was contained in a spherical tank separated by a rubber diaphragm from another tank containing gaseous nitrogen at a pressure of 2.4 atm. The mercury reserve for the neutralizer (0.5 kg) was similarly stored. The mercury displaced by the nitrogen entered the engine.

On 10 February 1970, ion engine No. 2 was turned on and operated continuously for 48 h. During that time, the engine characteristics in space were checked. The engine was then turned off, and on 14 February, ion engine No. 1 was turned on which operated for 3800 h with two short interruptions, until it failed on 23 July. The second engine, turned on as soon as the first one failed, operated continuously for 2011 h and

and failed on 17 October.

Each time, the activation of the ion engine began with the establishment of the discharge of the neutralizer, lasting about 4 min. For the next 1.8h, the engine was warmed up in order to degaz the working chamber, and during the next 50 sec a discharge was fired in the ionization chamber and was stabi- /48 lized for 1 minute. Thereafter, an accelerating voltage was applied, and a beam of ions from the ion engine simultaneously appeared.

In the course of the next 5 h, a careful monitoring of the operation of the engine and auxiliary equipment was carried out at thrust levels of 30 and 80% of the rated thrust, then the engine was switched to 100% operation. The parameters of the ion engine agreed very well with the laboratory values.

A bias potential relative to the SC potential could be applied to the neutralizer of each ion engine. This was done in the course of the tests. It was found that the SC potential of -20 V and dropped to -20 V at a zero bias potential. No radio noise was generated during the operation of the engine.

Tests of ion engines under the SERT-2 program confirmed the long service life and operational stability of the engines in space. As was later determined, failure of both engines was due to excessive erosion of the accelerating electrode near the neutralizer. The data of the space experiment proved invaluable in the design of new, improved and long-lived models of of ion engines, and in particular, of the above-described plasma-ion engine for the SEP program.

<u>Tests of plasma engines in the "Meteor" earth satellite.</u> Space tests of stationary plasma engines with azimuthal drift and an extended acceleration zone were first conducted on the Soviet earth satellite "Meteor", placed in orbit in late December 1971. The satellite was equipped with an electric propulsion unit (Fig. 15), which included two engine blocks (engine with main and backup compensator cathodes), a system of delivery and storage of the working medium (xenon), an energy conversion system, and a control system. One engine block was powered by solar batteries via buffer storage batteries. The engines were supposed to function at the rated power (400 W) /49 for no less than 100 h. In addition to the engine unit, the satellite carried meteorological and experimental instruments, and control and radio equipment. The engine unit was controlled by radio commands from earth.

The testing of the electric propulsion unit included the following objectives :

a) to check the operation of EPE in space;

b) to check the combined operation of the engine unit with the systems of the satellite, and the transmission of radio commands from earth;

c) to measure the thrust and other parameters;

d) prolonged tests in order to change the orbit of the satellite before it enters the intended synchronous orbit around the earth.

Figure 15. Space electric propulsion unit of the "Meteor" earth satellite. Shown are the engine blocks, a conversion and control system, and a system for supplying and storing the working medium.

The engine blocks were mounted on the outside of the satellite container on brackets: one in the direction of the course and one in the opposite direction. The jets were oriented at an angle of 4° (DB-2 in the opposite direction) and 5° (DB-1 in the course direction) with respect to the direction of motion of the satellite in order to control the thrust in its action on the satellite's attitude control system via the torsional moment. DB-1, which operated in the direction of the satellite's course, retarded and reduced it, and DB-2, which operated in the opposite direction, had the opposite effect. To increase the reliability in the conversion and /50 and control system, there were two channels (main and backup) providing for the operation of each engine block.

By successively turning on DB-1 and DB-2, their electrical parameters and the transmission of radio commands were checked. It was found that the engines did not affect radio communication.

On 2 February 1972, the solar battery panels were deployed at an angle of 20° with the course, the reverse side facing the plasma. Since the aperture angle of the jet issuing from the engines was 26°, this completely excluded the action of the plasma of the engines on the photocells. After the deployment, the thrust measurement was made; for this purpose, each engine block was turned on and operated for two loops.

On 14 February 1972, the engine unit was activated for prolonged operation and ran for 170 h. During the period from 14 to 22 February, the satellite was shifted by means of the plasma engines (DB-1 was operating) to an orbit close to the intended synchronous near-earth orbit, and daily, for 14 loops, crossed the equator at the same longitude. The height of the orbit during the operation of the engines changed by approximately 17 km. According to the data of trajectory measurements, the thrust of both engines differed little from the laboratory values and amounted to about 2 g.

Thus, the intended testing program was carried out successfully. The stationary plasma confirmed their reliability and workability under space conditions. For the first time in the history of EPE testing during the very first experiment, they were used to execute the proper correction of the orbit of an operating earth satellite.

Routine use of EPE. When suitable space power equipment becomes available, one should expect a marked increase in the number of launches of satellites equipped with space electric propulsion units, since EPE are eminently suited for attitude control, stabilization and orbital correction of near-earth /51 satellites. These maneuvers are indispensable for a successful and prolonged functioning of satellites whose orbit and orientation change slowly under the influence of various types of disturbances. Only a very slight force, or torsional moment, is required to orient a satellite as desired and to maintain the required orbital parameters. We shall mention only a few studies.

In June 1974, the American communications satellite ATS-6 was launched, carrying two cesium ion microengines for stabilizing the satellite's orbit in the north-south direction. The pulsed microengines, very similar to those tested on the LES-6 satellite and on the SMS-1 geostationary weather satellite launched in May 1974, will be installed in the LES-8 and LES-9 communications satellites (the launch is scheduled for 1975). The power consumed by the engine unit will not exceed 25 W. The objectives of the engine unit will include stabilization of the satellites in the east-west direction and correction of the orbit. Finally, it is intended to install four mercury ion engines 8 cm in diameter on the commercial communications satellite of the "Intelsat" consortium for stabilizing the satellite in the north-south direction and correcting the orbit. The "Intelsat" satellite weighs 900 kg and is designed for 10 years' operation. Solar batteries will be used as the power sources in all the projects.

Future of EPE

Space engines with a large specific impulse. Apparently, the creation of electric propulsion satisfying any reasonable sets of requirements is now possible. However, modern versions of high-power generating equipment lead to heavy and cumbersome designs. For this reason, studies are being conducted on a number of nonelectric devices using nuclear energy. Almost all are based on heating hydrogen to a high temperature, then allowing it to escape, i.e., are based on the use of the same nozzle /52 as in thermochemical engines [1] The most promising design is that of a heterophase nuclear reactor, with channels through which hydrogen, which is heated to a temperature of 2000-2500°K, is circulated. The calculated exhaust velocity is 8.5 km/sec. In the plasma phase reactor scheme, hydrogen should either "wash" a sphere of uranium plasma heated to a temperature of 50,000°K (in this case the calculated exhaust velocity can be raised

1 Analogous proposals based on the use of solar energy appear to be less promising and practically no studies of them are being made.

to 26-65 km/sec), or should be introduced into the zone of a controlled thermonuclear reaction (which has not yet been carried out) producing an even higher exhaust velocity[1]

Thermal nuclear engines will eliminate the need for a high-capacity on-board electric power plant. However, they still are cumbersome and very complex. It is necessary to minimize the danger of radioactive contamination of outer space and to ensure a high reliability of reactor operation at high temperatures. Hydrogen should be stored in liquid form aboard SC for months if not years. It should also be considered that liquid hydrogen is almost 15 times lighter than water, and hence, tanks for its storage must be very bulky. The existing laboratory models of nuclear rocket propulsors are very far from meeting the existing needs. It is therefore difficult to determine how the competition between the indicated schemes (electric [2] and nuclear) of high-power engines will evolve, particularly since the manned space flights for which they will be required will not begin until the end of the 1980's. However, the current impression is that time is generally in favor of EPE. It is therefore quite possible that the first manned interplanetary flight

1 Schemes involving no heating of hydrogen appear highly exotic at this time. For example, it has been proposed to accelerate a spaceship by exploding small atomic bombs in its vicinity.

2 With both solar and nuclear power equipment

will still take place with the aid of EPE. In any event, until the end of the 1980's, the only high-specific impulse propulsion units in actual operation will be the space electric propulsion units.

If however one turns from the relatively distant prospects to the more immediate future, the situation is as follows. /53 The power generating equipment aboard SC, still limited in power, will be generally designed primarily for the operation of equipment on board SC. On the other hand, this will considerably facilitate the adoption of EPE in space engineering, since in many cases, a space electric propulsion unit can operate during the "silence" periods fo the on-board instruments and thus, it will not be necessary to create a special energy source for EPE. On the other hand, such "nonautonomy" of a propulsion unit will result in the installation of only low-power EPE on board SC. This will inhibit the development of high-power EPE. Nevertheless, the process of adoption of EPE in space engineering has already begun and is cc .nuing successfully. "Nonautonomous" EPE are needed primarily where:

a) It is necessary to ensure a long existence of an SC whose orbit changes under the influence of various types of disturbances (for example, drag for low-flying SC). To maintain the orbital parameters, it is convenient to use engines with a high exhaust velocity, i.e, EPE, otherwise the reserve or the working medium becomes too large.

b) It is necessary to maintain an exact orientation or flight trajectory of the SC (for example, a space observatory). In this case, the usefulness of EPE is explained by their high dynamic characteristics and the possibility of generating very small, accurately controlled thrust pulses (electromagnetic oscillation of a beam in ion engines or pulsed operating mode of plasma engines).

Active preparation for a qualitatively different stage in the development of EPE, i.e., the creation of "autonomous" space electric propulsion units, has now begun. We have in mind, in particular, the American SEP (solar electric propulsion) program, under which an integrated series of SC with EPE is being developed. A special solar power equipement with a power of 20kW is being developed for them. The program is designed for the next 15 years and is expected to solve a wide range of problems. Some of them will be discussed here.

Optimality criteria of "autonomous" space electric propulsion units. Each SC has a program, which represents the /54 purpose of the flight. The objective of SC designers is to come up with an optimum design of this device and to provide for its optimum operation in order to solve the stated problem (or series of problems).

One on the most important optimizable elements is the engine unit. Since the operating conditions of engines in different SC are very different, there is of course no optimum engine for all the cases that may arise. Nevertheless, certain optimality criteria can be formulated for a certain range of problems.

We shall consider two problems involving optimization of an autonomous electric propulsion unit. We shall begin with the simpler one.

Optimum specific impulse. Let us consider a flight during which the engine unit should develop a given constant thrust F during a given time τ . We shall also assume that the thrust efficiency η_F and the "specific mass" K of the SC power plant, usually measured in kilograms per kilowatt (the mass of the power plant u_p is proportional to its power: $u_p = u_1 + KP$), are known. The question is, what should the exhaust velocity be in order to minimize the takeoff mass of the propulsion unit?

The fact that at a certain exhaust velocity the mass of the space electric propulsion unit should be minimum follows from some very simple considerations. Actually, at a low exhaust velocity (approaching zero), the production of the necessary thrust F involves a large mass flow rate of the working medium, so that it will be necessary to take along a large reserve of this medium. In the other limiting case, when the exhaust velocity is very high, the production of the necessary thrust will involve a very large energy consumption, i.e., the mass μ_p of the energy source will increase. It is clear that the optimum exhaust velocity is somewhere in between these two extremes. A simple calculation shows that the mass of the propulsion unit is minimum at the following exhaust velocity: $u_{opt} = \sqrt{2\eta_F \tau / K}$. It is evident that when $K = 10$ kg/kW, $_F = 50\%$ and $= 1000$ h, the optimum exhaust velocity is 20 km/sec.

Optimum space flight with constant power, The above /55 formula for the optimum exhaust velocity at which the minimum mass of the electric propulsion unit is reached applies in the case of a strict and artificial assumption of a constant thrust, and it is therefore generally suited only for estimates. It is more realistic to assume that the engine continuously consumes a constant power P, although at different times the thrust (and hence, the exhaust velocity) may change over arbitrarily wide limits. Thus, we are dealing with an optimum flight with an ideally controllable engine.

Using the formulas for the thrust and thrust power, one can obtain the following formula relating the final mass μ_f to the initial mass μ_0: $\mu^{-1}_f = \mu^{-1}_0 + Y/2P\eta_F$, where $Y = a^{-2}\tau$ is the time-averaged square of SC acceleration under the influence fo the engine unit alone multiplied by the flight time. It is readily apparent that at a given thrust power $P\eta_F$, the minimum mass flow rate is obtained when Y is minimum. This minimum is associated with completely defined flight trajectories and engine thrust control conditions in the course of flight. Actually, a flight from point A (for example, earth) to point B (for example, Mars) in time τ can take place along the fanciest trajectories, but only along one (or a few) trajectories will Y be minimum. There exist some completely defined (although very complex) rules for finding the optimum trajectories and mode of optimum control of the thrust.

Having the mass formula and knowing the minimum value of Y (which will be denoted by Y_{min}), one can find the power of a plant for which the takeoff weight of the SC will be minimum for a specified payload.

As an example, we shall consider a space flight from earth to the orbit of Mars. Figure 16[1] shows the lines of constant Y values and lines of maximum accelerations a_{max} in the coordinates: date of launch t_1 vs. date of arrival t_2 /56
It is obvious how the flight time can be sharply reduced by choosing an optimum launch date and increasing a_{max}. For example, if one takes as the launch date 28 September 1960 and $Y = 11.54\ m^2/sec^3$, then, at an efficiency of 70% and with K = 13 kg/kW, it turns out that the fraction of the payload will be 45 % of the takeoff weight. An estimate of the optimum exhaust velocity in this case gives a value of 44 km/sec, i.e., a specific impulse of over 4000 sec.

1 Let us emphasize once again that we are dealing with the acceleration of an SC under the action of the engine unit alone along the actual flight trajectory, determined by the combined action of the reactive thrust and other external force (mainly gravitation)

The proportion of the payload decreases sharply with increasing K, and when K = 100 kg/kW, it turns out to be less than 1%.

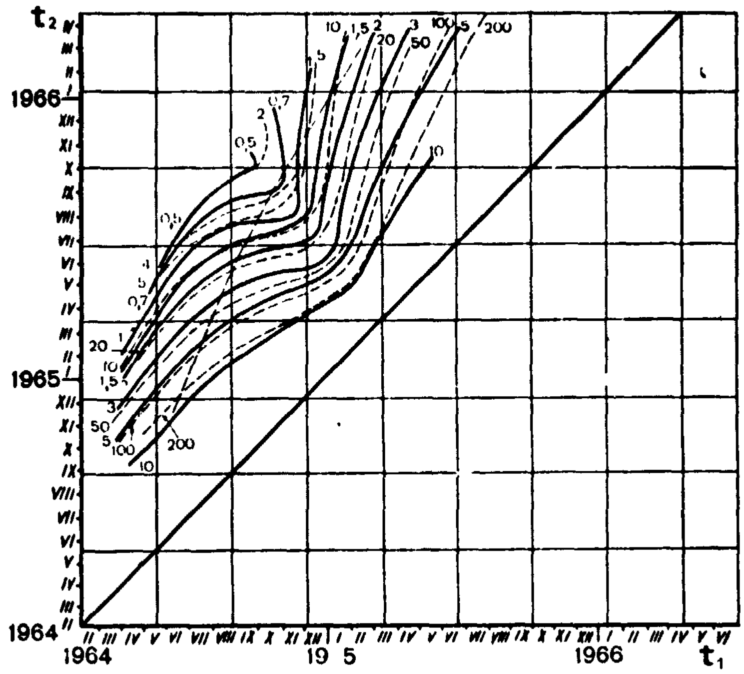

Figure 16. Isolines of characteristics of the flight to Mars. Solid lines - isolines of maximum accelerations [mm/sec^2]; dashed lines - isolines of Y values [m^2/sec^3]

The condition of constant P is characteristic of ideally adjustable engines with a low thrust and an external energy source. There are also situations in which the exhaust velocity is constant, and the power is not limited in principle. This case occurs in flights using chemical rocket engines / 57 (for example, LRE). Since the corresponding mass formula is Tsiolkovskiy's formula, the characteristic velocity $v = \bar{a}\tau$ is to be optimized (minimized).

Electric space shuttle. In the region from low orbits to the geostationary orbit, a complex of orbital stations, laboratories, and satellites of diverse functions has been created and is steadily growing. A successful operation of this "multistory" complex makes it necessary to provide for a durable and trouble-free operation of the spacecraft. In addition, it is necessary to make the space launch process itself cheap and universal (since the number of launches increases every year), with the largest possible payload fraction. These problems can be successfully solved by using high thrust rocket engines and EPE.

The carrier rockets used at the present time are systems used only once. During the launch phase, the spent stages of the rocket separate from the SC and burn in the lower layers of the atmosphere. Each launch is very expensive. Therefore, in order to deliver a load to a low near-earth orbit with a height of 200-400 km, a transportation spacecraft (TSC) is now being intensively developed, i.e., a multiple-use vehicle. The TSC is a superhigh-altitude winged cruise-type rocket that is placed in orbit by means of chemical rocket engines. The TSC lands on the earth like an ordinary airplane. It is expected that each TSC will be used at least 100 times, lifting about 20 t of load to a low orbit in each launch. The use of TSC for lifting loads to heights greater than 200-400 km is much less desirable: the mass of fuel for its engines increases, and the proportion of the payload decreases. At the same time, the use of low thrust engines (EPE) for transporting loads

directly from a low orbit to a geostationary one is not always efficient either. This is because, starting at a height of approximately 500 km to 14,000 km, the earth is surronded by so-called radiation belts consisting of energetic charged /58
particles trapped by the earth's magnetic field. If it is desired to transfer an SC to a geostationary orbit, to avoid radiation damage to the instruments, it is best to traverse the region of the radiation belts as fast as possible by using high thrust engines. For this purpose, TSC are used to lift to a low orbit a special multiple-use rocket towing stage, operating on an oxygen-hydrogen mixture and delivering loads from a low orbit to an intermediate orbit(14,000 km)and back. EPE can already be used to transport space objects from an intermediate orbit to a geostationary one. We shall consider two projects involving the creation of an "electric space shuttle" (ESS).

According to the project of the British company Rolls Royce, the energy source for the ESS is a nuclear reactor, and the EPE used are either electrically heated engines operating on hydrogen, or plasma propulsors with azimuthal drift, operating on rare gases (for example, argon). Because of the high specific impulse of EPE (in this case, from 800 to 1500 sec), the proportion of the payload transported by means of ESS increases substantially. Thus, for a 20-ton takeoff weight of the rocket in a low orbit, an ESS with argon engines lifts over 13 tons of load to a geostationary orbit (an ordinary rocket lifts 6 t), and the cost of the lift is 1/4 of that involved in the use of a chemical rocket.

ESS[1] with solar batteries is a component part of the American SEP project. This project stipulates the creation of universal rocket stages with electric thrust for near-earth and interplanetary flights. The thrust is produced by the above-described mercury ion engines with a diameter of 30 cm and a specific impulse of 3000 sec. The ESS for near-earth flights has a length of 3m; 9 propulsors are mounted in its rear, and a special system for docking with the space objects being towed is mounted in the front. Two solar panels (13 kg/kW) generate 25 kW or electric power, 21 kW of which is supplied to the engines. The efficiency of the electric propulsion system is 66%. Not more than 7 engines operate simultaneously. The weight of the ESS is 2700 kg, 1500 kg of which corresponds to the mercury reserve, /59
contained in four tanks. This amount of mercury is sufficient for 530 days of operation of the ESS engines (the engine service life being 10,000 h). By using the ESS-tug-TSC combination, in 180 days, one can lift 6 t on load to a geostationary orbit, return 5.5 t of load from a geostationary orbit, or simultaneously provide for the lifting and

1 Pictured on cover.

return of 3.5 t of load. Let us note for comparison that the TSC - tug combination is capable of operating with a much smaller load (depending on the specific type of tug, the weight of the load turns out to be 2-6 times smaller'). The use of ESS is convenient and economical: for example, while the launching of a geostationary communications satellite of the "Intelsat" consortium by an ordinary rocket costs $15 million, the use of electric thrust makes it possible to reduce this cost by a factor of over 2.5!

The SEP project also stipulates the creation of a special service and repair satellite. Its propulsion system consists of 4 ion engines, two of which operate simultaneously. The solar batteries generate 7 kW of electric power, 5.5 kW of which is supplied to the engines. The satellite weighs 350 kg (100 kg corresponds to the mercury reserve). It is equipped with a television camera and telemanipulators.

The service satellite approaches a satellite in a geostationary orbit, surveys it, and, upon finding the malfunction, performs the repairs on the spot. If however such repairs are impossible, the service satellite can tow the malfunctioning spacecraft downward, to an intermediate orbit, then return it. Such a satellite is designed for several years of operation.

It is expected that near-earth flights under the SEP project in conjunction with TSC will begin in the early 1980's.

Interplanetary flights under the SEP project, using ordinary carrier rockets (of Titan-Centaur Type) in the early stages, are scheduled to begin in the later part of the 1970's.

Interplanetary flights and flights to comets and asteroids.
We have already stated that the use of high-specific-impulse
EPE for space flights will permit a considerable increase in
the proportion of the payload carried by a spaceship. However,
this is not the only advantage of engines with electric thrust.
When operating under weightless conditions for a long time, they
are capable of accelerating a ship to high velocities or exe- /60
cuting complex maneuvers with a large change in characteristic
velocity.

What projects involving interplanetary flights by means of electric thrust are being actively worked on at the present time?

The SEP project stipulates the creation of a special universal electric rocket stage for accelerating a spaceship from an intermediate orbit. This stage closely resembles the above-described electric space shuttle. The propulsion system includes 8 ions engines (maximum number of operating engines, 6) powered by 21 of the 25 kw generated by the solar panels. The weight of the stage is about 1 t (excluding the mercury reserve, which varies for different flights). The parameters of the stage are chosen so as to provide for an effective acceleration of different unmanned ships or scientific equipment.

Under the SEP project, the most important flights are :

a) slow flyby of Encke's comet (1979);

b) Jupiter-Uranus flight (1979);

c) flight to Jupiter with capture in orbit around it (1980);

d) encounter with Encke's comet (1981);

e) flight to Venus for the purpose of mapping it by means of radar (1985);

f) Uranus-Neptune flight (1986);

g) flight to Mercury with orbital capture (1987);

h) flight to Saturn with orbital capture (1989);

i) encounter with the asteroid Metis (1989);

We shall examine some of these programs in more details.

The launch of the flight to Encke's comet with a slow flyby is scheduled for early January 1979, and the flight time will be 670 days. The space load includes 113 kg of scientific equipment and a block of service systems weighing 420 kg, necessary for the operation of the scientific instruments, monitoring, radio communication, etc.

The launch of a satellite to Jupiter is scheduled for the end of November 1980. The engines of the electric rocket will operate continuously for 400 days; during the next 500 days the ship will travel by inertia. The final weight of /61 of the SC together with the block of service systems will be approximately 1.3 t. The use of a braking rocket stage is proposed for the orbital capture.

The flight to Mercury (launch, May 1987; arrival, August 1988) will last 450 days, during which the EPE will operate continuously. A braking stage is also used for capture in an orbit around the planet. The weight of the SC and block of service systems will be 1 t.

In addition to the flights mentioned above, a variety of other expenditions to the sun, comets and asteroids have been planned. At the present time, it is difficult to state precisely what flights be carried out under the SEP project. We have discussed this project in detail because its technical implementation is no longer in doubt even day. There are many other space projects involving the use of EPE, which we shall not consider because they have no been sufficiently studied.

Conclusion

An intensive development of EPF began about 15 years ago. During that time, a very long road has been traveled, from very vague ideas to electric propulsion systems with unique characteristics, tested and successfully operating in space.

Modern EPE make it possible to obtain an exhaust velocity above 100 km/sec. Their service life is as long as 10,000 h, and the number of cut-ins is measured in tens of thousands (for stationary EPE) and hundreds of millions (for pulsed EPE). These engines are simple from both the structural and the operational points of view; they have a wide range of autonomous adjustment of the main parameters (thrust and exhaust velocity) without an appreciable change in efficiency. In ion and electromagnetic engines, the direction of the thrust vector can be controlled without any mechanical devices. EPE can operate both steadily and in short pulses (on the order of milliseconds and microseconds). The power level in modern single stationary EPE modules varies from tens of watts (electrically heated, ion microengines) /62 to hundreds of kilowatts (end-type-engines [1]), and thus far, no limit is in sight for shifting to low or high powers.

In principle, the advantages enumerated above can make EPE the ideal rocket engines. Progress is somewhat slow because the existing on-board electric power sources are very complex, heavy, and have a low power. However, their improvement is making rapid progress.

In the final analysis, there are several simple but fundamental factors which assure the EPE a great future.

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1 And quasi-steady end-type engines operating in millisecond pulses consume a power of the order of megawatts! However, even this figure is negligibly low in comparison with the power of modern carrier rockets, once again confirming the' inevitability of retaining thermochemical rocket engines for a long time to come as the only ones suited for lifting SC from the surface of earth (or other planets) and their landing.

They are as follows:

1. Thermochemical and solid phase nuclear rocket engines are unable to provide the optimum high specific impulse required for many tasks.
2. In its simplicity, versatility, controllability, service life and reliability, EP should surpass classical low power propulsion systems [2] in the near future.
3. The development of space (particularly solar) power engineering is proceeding at a very rapid rate independently of the development of EPE. Thus, EPE benefit by receiving as a "gift" the chief prerequisite for their existence - on-board electric power equipment.
4. The development of methods of generating powerful ionic and plasma fluxes is also proceeding largely independently of EPE. These fluxes are necessary in physical laboratories for research in controlled thermonuclear fusion, for vacuum technology, etc. This signifies that the physical base of EPE will itself be steadily improved. All of this taken together indicates a limitless and inexorable adoption of EPE in the space technology of the future.

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